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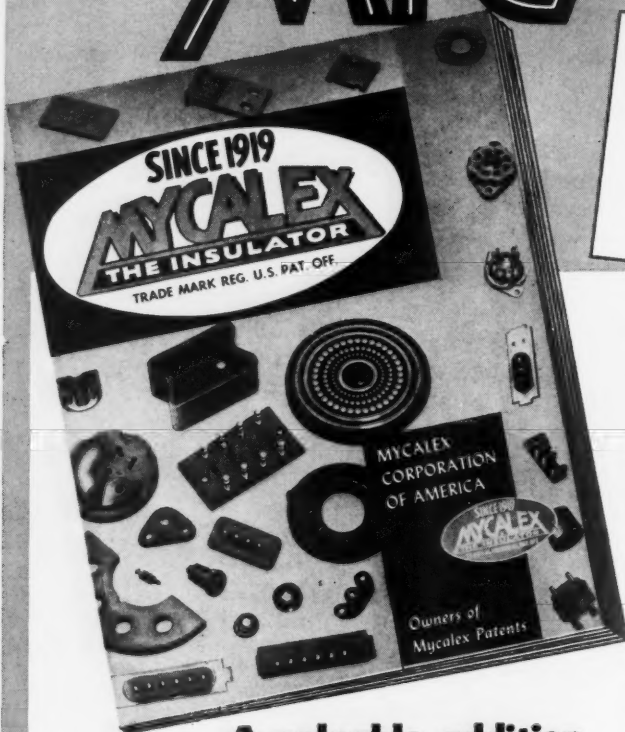
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Equipment for Use with High-Strength Hydrogen Peroxide

By NOAH S. DAVIS, JR.,¹ and JOHN H. KEEFE, JR.²

Buffalo Electro-Chemical Company, Inc., Buffalo, N. Y.

A search was made for the most satisfactory equipment for handling concentrated H_2O_2 in storage and propellant systems. The results of compatibility tests on various materials, including metals, plastics and lubricants are presented. General equipment design considerations and methods of preparing the equipment for H_2O_2 service are outlined. Examples of the following equipment are presented and discussed: Pumps, containers, piping, flexible hose, valves, and reactors.

I Introduction

HIGH-strength hydrogen peroxide is today in wide use as a monopropellant and as an oxidant in bi-propellant rocket systems. In World War II the Germans developed numerous military applications for high strength H_2O_2 , such as the Messerschmitt ME 163-B rocket fighter, the Focke-Wulf ATO FW 56, and the Henkel He 112 and He 126 rocket motors.³ Since the end of the war there have been continuing developments, both in America and abroad, in the utilization of hydrogen peroxide. Most of these developments have been military and of necessity are under security classification. However, out of these developments have emerged several considerations which form the basis for the sound design of equipment for handling H_2O_2 .

In this paper these design concepts will be discussed and examples will be given of items of equipment embodying these principles.

The physical properties of high strength H_2O_2 and its performance as a rocket propellant were discussed in a paper presented by R. Bloom, N. S. Davis, Jr., and S. D. Levine at the December 1949 meeting of the American Rocket Society (2).⁴

Presented before the AMERICAN ROCKET SOCIETY at Toronto, Canada, on June 11, 1951.

¹ Manager, Special Projects Department. Member ARS.

² Project Supervisor.

³ In this country, Lt. Col. F. Bellinger and associates, of the Chemical Warfare Service, carried out numerous experiments (1)⁴ using hydrogen peroxide as a propellant. In this work, many of the materials of construction were evaluated.

⁴ Numbers in parentheses refer to Bibliography on page 69.

II General Considerations in Selecting Equipment for H_2O_2 Service

A Material Selection

The first and most important consideration in choosing or designing equipment for H_2O_2 service is that of materials selection. It is essential that all parts of the apparatus which will be exposed to the H_2O_2 be made of sufficiently compatible materials. Extensive tests have been made to determine which materials should be used. The decision as to the compatibility of a certain material was based on the following factors:

1 The effect of the material on the rate of decomposition of the H_2O_2 —numerous metals, such as lead and metallic oxides, or salts, such as lead oxides used in pipe dopes, are effective catalysts for the decomposition of H_2O_2 . Their use is dangerous and must be avoided.

2 The effect of the H_2O_2 on the materials—certain materials such as thiokol rubber burn on contact with H_2O_2 , and others such as the 400 series stainless steels rust.

3 The possibility of forming detonable mixtures with H_2O_2 —combustible liquids such as ethyl alcohol form mixtures with high strength H_2O_2 which have the sensitivity and bursting power of nitroglycerine.

Laboratory tests were made to determine the effect of the peroxide on the sample and the effect of the sample on the peroxide. In these tests, strips of the material were immersed in 90 per cent H_2O_2 for a set time at a controlled temperature. The rate of decomposition of the solution was measured to show the effect of the sample on the H_2O_2 . The sample was observed after exposure to the H_2O_2 to show the effect of the H_2O_2 on the sample. The impact sensitivity of H_2O_2 containing mixtures was determined in a modified Bureau of Mines drop weight tester.

In deciding which materials should be used for a particular item of equipment, the end use of the item must be a dominant factor. For example, a long-time storage tank should be made of highly compatible material, whereas a valve in a one-shot rocket could be made of less compatible material. Materials have been

divided into four general classifications depending on their end use.⁵ In Table I are shown the results from tests on some of the materials which were evaluated.

Class 1: Materials which are highly compatible and can be used for long-time contact with H_2O_2 . Irrespective of length of contact, these materials will not contaminate the H_2O_2 . Typical use for a material of this class would be for the construction of long-time storage containers such as tanks, tank cars, or drums. Pure 99.6 per cent aluminum and Teflon are examples of this class of materials.

Class 2: Materials which are satisfactory for repeated short-time contact with high strength H_2O_2 . These materials can be used for transient contact with the solution prior to storage of the H_2O_2 or for limited contact of the H_2O_2 prior to prompt use. Such limited contact time should not exceed 4 hours at 160 F or 1 week at 70 F. Typical uses for materials of this class would be as valves or pumps in a peroxide transfer line, or as high-pressure storage tanks. The following materials typify those satisfactory for Class 2 service: Aluminum alloys 43, 52S, 61S, the AISI type 300-series stainless steels, and certain polyvinylchloride plastics such as Vinylite or Koroseal.

Class 3: Materials which should be used only for short time contact with H_2O_2 prior to prompt use of the H_2O_2 . These materials can be used for repeated contact with H_2O_2 , but no one period should exceed 1 minute at 160 F or 1 hour at 70 F prior to immediate use. These materials might cause sufficient contamination of the solution to render it unsuitable for storage. Some of the materials in this class are Hastelloy B and C, Refractalloy 26 and 70, and Universal Cyclops 19-9DL. A typical use for materials of this class would be as a valve in a one-shot H_2O_2 propulsive unit.

Class 4: Materials which are not recommended for any use with high strength H_2O_2 . They are materials which cause immediate excessive decomposition of the H_2O_2 , are rapidly attacked by the H_2O_2 , or form explosive mixtures with the H_2O_2 . Typical examples of this type of material are copper, lead, 400-series stainless steels and numerous other alloys, plastics, and lubricants, including the silicones.

The chemical composition of a material is not the sole determining factor as to whether it is compatible with H_2O_2 . The liquid phase decomposition of H_2O_2 , apparently, is predominantly a heterogeneous reaction and is, therefore, markedly affected by the nature of the surface exposed to the H_2O_2 . In general, the smoother the surface the less will be the rate of H_2O_2 decomposition. Type 300-series stainless steels in the wrought form or with a machined surface are relatively compatible. The same metal alloys when used with a rough cast surface are not compatible. They cause excessive decomposition of the H_2O_2 . At the same time, the stability of the solution is decreased because chro-

⁵ This method of approach differs from that used by Lt. Col. Bellinger, in which analytical results were reported and the reader was required to decide whether the material was suitable for a given use.

TABLE I. CLASSIFICATION OF MATERIALS ACCORDING TO RECOMMENDED USE WITH 90 PER CENT H_2O_2

Material	Class 1: Long time contact	Class 2:	Class 3:	Class 4: Unsuit- able
		contact prior to stor- age	Short time contact prior to use	
<i>Metals:</i>				
Aluminum—99.6% pure	X	X	X	..
2S Alloy	X	X	X	..
43	..	X	X	..
52S	..	X	X	..
56S	..	X	X	..
61S	..	X	X	..
63S	..	X	X	..
75S	X
150S	..	X	X	..
24S	X	..
13	X
40E	X
Copper	X
Lead	X
Iron or carbon steel	X
<i>Stainless steel AISI type</i>				
303	..	X	X	..
304	..	X	X	..
309	..	X	X	..
310	..	X	X	..
316	..	X	X	..
317	..	X	X	..
318	..	X	X	..
321	..	X	X	..
322	..	X	X	..
347	..	X	X	..
440	X
443	X
446	X
Durimet 20	..	X	X	..
Hastelloy B and C	X	..
Inconel	X
Monel	X
Refractalloy 26 and 27	X	..
Stellite No. 6	X
Universal Cyclops 19-9DL	X	..
<i>Plastics:</i>				
Polyethylene	X	X	X	..
Teflon	X	X	X	..
Kel-F	X	X	X	..
Buna S & N	X
Geon 8372	X
Hycar	X
Koroseal	..	X	X	..
Neoprene	X
Nylon	X
Polystyrene	..	X	X	..
Silicone Rubber	X
Thiokol	X
Trithene	X	X	X	..
Tygon 2807	X
Vinylite	..	X	X	..
Veloform	..	X	X	..
<i>Lubricants:</i>				
Aroclors	X
Fluorolubes	X	X	X	..
Kel-Flo Polymers	X	X	X	..
Paraffin	X
Perfluorolube Oils	X	X	X	..
RPM Hydraulic Fluid	X
Silicones	X
Ucon Hydrolube U-4	X
Skydrol	X
Halocarbon Oils	X	X	X	..

NOTE: This table is not complete. For additional recommendations, see Reference (3).

mium is readily leached from the cast surface.

Certain surface treatments such as electropolishing improve the compatibility of "poor," that is, relatively incompatible alloys by smoothing the surface. Other surface treatments, such as nitriding to improve hardness, render otherwise compatible stainless steels incompatible.

Approximately thirty plastic or protective coatings were tested with 90 per cent H_2O_2 . Of these, the completely halogenated polyethylenes, Teflon and Kel-F, were found to withstand the action of H_2O_2 up to temperatures of approximately 160 F, and polyethylene to 120 F.

In general, platings, such as tin plate, strip off when exposed to 90 per cent H_2O_2 . Apparently, the H_2O_2 seeps through pinholes in the plate or under the edge. Once the H_2O_2 contacts the undersurface it decomposes, liberating oxygen gas 400 times the liquid volume. The gas forms a blister which eventually breaks and allows more H_2O_2 to contact the undersurface.

As a general rule, only those materials and surface treatments should be used which have been found to be compatible with H_2O_2 .

B Design Considerations

Having decided on the proper materials to be used in fabricating a certain piece of equipment, the next consideration should be given to various design practices. In general, the same principles of design apply to equipment for H_2O_2 as are customarily practiced for other equipment. That is, the thickness of the walls should be based on pressure requirements, the shape of the port opening in throttle valves should be calculated to give the desired flow characteristics, and good welding techniques should be practiced. In addition to these, the following specific points should be remembered for the design of equipment for H_2O_2 service.

1 There should be no places in the equipment where peroxide can be trapped. H_2O_2 solutions will constantly decompose at a slow but steady rate, and if completely confined, the resulting gas will eventually build up sufficient pressure to cause rupture. A plug is an example of a design where the peroxide can be entrapped. When the valve is in an open position, peroxide will fill the hole in the plug and will be trapped within the plug when the valve is closed. On the other hand, a globe valve is better suited for peroxide service because, in general, there are no points where peroxide will be trapped.

2 The valve, pump, or other pieces of equipment should be designed so that it can be completely drained on shutdown. This is desirable so that when not in use the system will contain no stagnant pools of peroxide.

3 Dead ends should be avoided. By dead ends are meant those places which could be filled with H_2O_2 without permitting adequate recirculation of the fluid. An example of this is a typical Bourdon tube pressure

gage. The disadvantage of having a dead end in a piece of apparatus is that there is a possibility that small impurities will be swept into the dead end and there accumulate until extensive decomposition of the H_2O_2 results.

4 The restriction as to the types of material which can be used with high strength H_2O_2 can sometimes give rise to difficulties due to galling of mating metal parts having the same relative hardness. AISI Type 300 series stainless steels and soft aluminum alloys are especially bad in this respect. Therefore, precautions should be taken to keep to a minimum those parts requiring the use of mating 300 series stainless steel or aluminum threads. Ample clearance between the threads should be provided as in a Class 1 fit and a gasket used for sealing. In addition, the threads should be coated with a compound to prevent seizing or galling. Most, if not all, of the usual antiseizing compounds are not compatible with H_2O_2 . However, tests have indicated that the polyethylene cement, Dispersite 1822-A, made by the Naugatuck Chemical Company, is satisfactory for this use. After about one month's contact with 90 per cent H_2O_2 , the Dispersite will become soft and tacky. For equipment to be used in indefinitely long contact with H_2O_2 , the threads should be coated with polyethylene. This can be done by applying granulated or rod polyethylene to the male threads which have been heated to about 150 F. The coated part should then be quenched in cold water and screwed into the female part.

5 Dissimilar metals in contact in the presence of 90 per cent H_2O_2 show a slight tendency for electrolytic corrosion. It is interesting to note that the more concentrated the solution the less is the galvanic action. However, even with 90 per cent H_2O_2 , it is good practice to avoid, where possible, the contacting of dissimilar metals. This can sometimes be done by insulating one metal from the other by a plastic. If two dissimilar metals must be in contact, the anodic of the two should have a larger surface than the cathodic.

6 In using welds to join two sections of a piece of equipment, care should be taken to avoid weld splatter on the inside of the equipment. It is best practice to do the welding by filling from the outside and using a backing ring made out of compatible material, or other method to keep to a minimum the weld exposed to the H_2O_2 . Stainless steel weld material which is exposed to H_2O_2 should be machined smooth.

C Preparation of Equipment for Service

Prior to placing a piece of equipment in H_2O_2 service, the parts should be cleaned and properly prepared. Plastics, in general, can be prepared by scrubbing them with a synthetic soap solution. Aluminum and aluminum alloys should be degreased by scrubbing them with a detergent solution, then treated with a dilute solution of sodium hydroxide and then with sulphuric or phosphoric acid and finally rinsed with distilled water. Stainless steel should be degreased and passivated with nitric acid solution. It is good practice

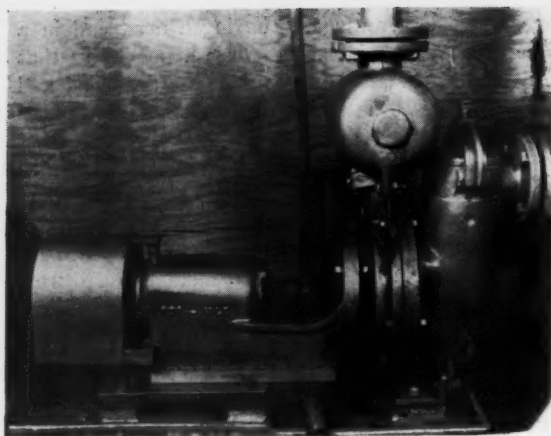


FIG. 1 LABOUR CENTRIFUGAL PUMP

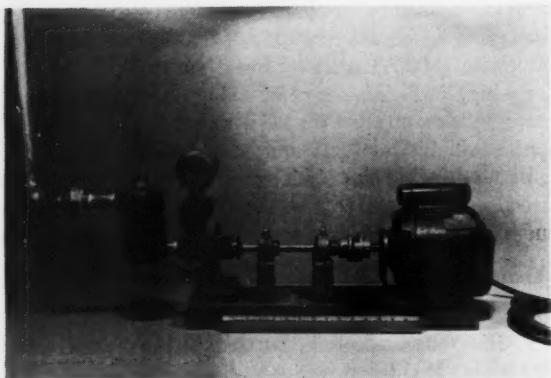


FIG. 2 PORTABLE CENTRIFUGAL PUMP

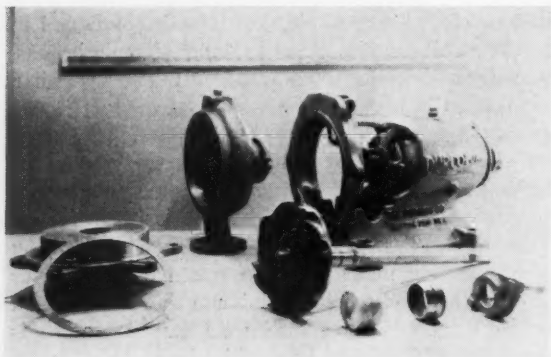


FIG. 3 DURIRON CENTRIFUGAL PUMP

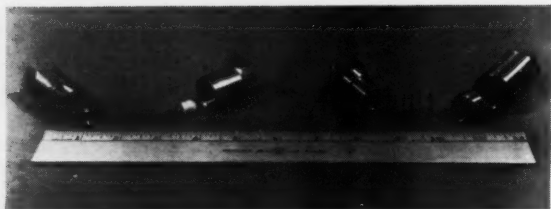


FIG. 4 1/4-IN. STAINLESS STEEL CHECK VALVES

to pretest with diluted H_2O_2 all items of equipment prior to placing them in service.

III Examples of Equipment

In the accompanying illustrations, various items of equipment are shown which illustrate some of the design concepts which have been discussed.

Fig. 1 shows a LaBour, cast aluminum-alloy 43 centrifugal pump which has a capacity of 35 gpm at a 70-foot head. This pump is equipped with a suction chamber and de-aerating chamber so as to be self-priming. Hydrogen peroxide solutions always contain a certain amount of dissolved oxygen which can sometimes vapor-bind a nonself-priming pump. LaBour pumps have been used extensively in peroxide service and have shown uniformly satisfactory performance.

Fig. 2 shows a small cast aluminum-alloy 43 centrifugal pump made by the Portable Pump Company. This pump has a capacity of 5 gpm at a 12-ft head. It also is self-priming. Braided Teflon is used for packing. This pump has an open impellor.

A third pump which has been used satisfactorily in handling 90 per cent H_2O_2 is the Duriron Company's Durcopump, Fig. 3. This pump has a capacity of 35 gpm at a 34-ft head. It is not self-priming. In use, it was found necessary to add a suction chamber to avoid loss of prime on start-up.

The Durcopump is equipped with a Teflon Duraseal. The seal is a Teflon bushing which is forced against the wear surface by the follower to prevent leakage along the shaft. At the same time, the conical end of the follower spreads the feather edge of the Teflon bushing to form a seal against the wall of the packing housing. One of these seals gave satisfactory service for a six-month period.

Other types of suitable pump and valve packings are: rings of braided Teflon packing impregnated with Fluorolube grease (Fluorolube is one of the completely halogenated hydrocarbons which is satisfactory as a lubricant in H_2O_2 service), shredded Teflon, and molded Teflon packings. A procedure (4) which DuPont developed for packing a valve with Teflon is to impregnate the metal parts with a fluorinated hydrocarbon. This can be done by thoroughly degreasing the metal, applying the fluorinated compound, inserting the packing, and then warming the entire valve to about 100 C. The fluorinated compound is adsorbed by the metal to produce a lubricated surface. It also is partially absorbed by and plasticizes the Teflon.

Tank farms and railway tank cars for the storage and shipment of hydrogen peroxide are made of pure 99.6 per cent aluminum, Heliarc-welded with pure aluminum rod. In large 25,000-gal capacity tanks, H_2O_2 has been stored with no detectable loss in concentration over a one-year period. Aboard missiles, H_2O_2 can be stored in fuel cells made of flexible plastics. The solution can be expelled from the container by applying outside pressure to collapse the cell. One of the plastics used was Vinylite VU 1930,

a polyvinyl chloride-polyvinyl acetate plastic. This plastic material turns an opaque white after approximately one week's exposure to 90 per cent H_2O_2 . After one or two months' exposure, the plastic is swollen and distorted and its tensile strength appreciably reduced. Hydrogen peroxide solutions which have been stored in polyvinyl chloride plastics pick up chloride ions and can corrode aluminum parts. For these reasons, polyvinyl chloride containers or gaskets are not ideal for long-time contact with H_2O_2 . Polyethylene does not have these disadvantages. Polyethylene suffers no apparent change in physical properties even after long-time exposure to H_2O_2 . The rate of decomposition of the peroxide is as low as when pure aluminum is used. Other storage vessels have been used satisfactorily with 90 per cent H_2O_2 , including several Type 347 stainless-steel tanks for the high-pressure short-time storage of H_2O_2 prior to immediate use. For an airborne, high-pressure, short-time storage tank, aluminum alloy 61S-T6 should be considered.

Figs. 4 to 10 show several valves, some of which are stock items while others are special valves especially developed for H_2O_2 service. Two spring-loaded $1/4$ -in. check valves are shown in Fig. 4. At the left, is a Jet Specialties valve with a Teflon plastic pintle; that at the right was made by the Kohler Co. and has a metal-to-metal seat. Both valves are manufacturers' stock items.

A 2-in. packless bellows valve, Fig. 5, was custom built for H_2O_2 service by the Fulton Sylphon Division of Robertshaw Fulton Controls Co. This valve has a confined Teflon disk which seats against stainless steel. It should be noted that all of the interior welds have been ground smooth. The bellows and all metal parts are of Type 347 stainless steel.

Fig. 6 shows a 300-series stainless steel valve especially designed for handling 90 per cent H_2O_2 . This was made by the Marotta Engineering Co. and is a 2-in. remote-control manual override valve. It is equipped with three seals so that a noncompatible hydraulic oil may be safely used as the actuating fluid. A seal at the top prevents leakage of the oil down the shaft; a seal near the bottom prevents leakage of the H_2O_2



FIG. 5 FULTON SYLPHON 2-IN. PACKLESS WYE VALVE

up the shaft. The lower weep hole is provided to drain off H_2O_2 just in case the seal does leak. An upper weep hole, not shown in the photograph, drains off oil leakage. A third seal located between the two vent holes effectively separates the two leak-off reservoirs to eliminate any possibility of the two fluids coming in contact to form a detonable mixture.

In general, large valves of sizes greater than 1 in. should have flanged connections. Threaded connections may be satisfactory for smaller valves provided that a compatible pipe dope is used. As cited previously, polyethylene or Dispersite 1822-A are satisfactory for sealing threaded connections.

Fig. 7 shows a 1-in. Hammel-Dahl throttle valve. The wet end was machined from a solid 347 stainless steel forging. It has a parabolic plug for flow characteristics and a Teflon ring for tight shut-off.

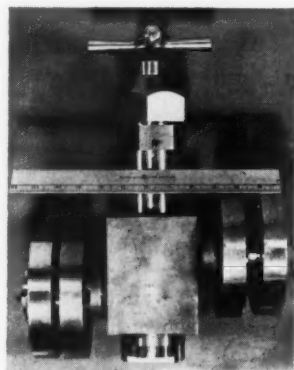


FIG. 6 MAROTTA 2-IN. HYDRAULICALLY OPERATED VALVE

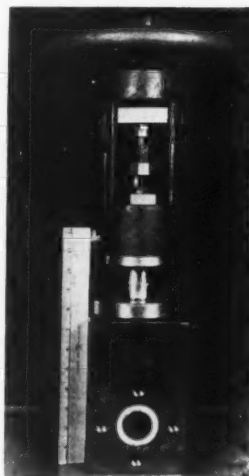


FIG. 7 HAMMEL-DAHL 1-IN. THROTTLE VALVE

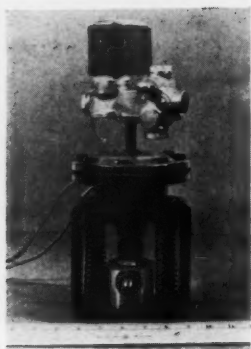


FIG. 8 ASCO 1/2-IN. PNEUMATIC SOLENOID VALVE

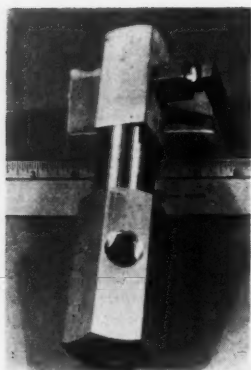


FIG. 9 FARRIS 1/2-IN. SAFETY RELIEF VALVE

Fig. 8 shows an Asco combination pneumatic solenoid valve. This valve is normally closed. When the solenoid is tripped, it opens an air valve to supply 90 psig pneumatic pressure to open the H_2O_2 valve. When the current to the solenoid is interrupted, or the air pressure fails, the H_2O_2 valve closes. The H_2O_2 valve is suitable for 1000 psig service.

Fig. 9 shows a Farris 1/2-in. safety relief valve. It is necessary to use a bursting disk or safety relief valve in any system where H_2O_2 might inadvertently be completely confined.

Fig. 10 shows two 1/2-in. Alloyco needle valves of forged 300-series stainless steel and with Teflon seats. A plastic-to-metal seat is desirable, especially in needle valves. Metal-to-metal seats of 300-series stainless steel are subject to galling. The valve at the left is equipped with an outside yoke for actuating the stem. This construction is superior to that of the valve on the right because the bushing then does not contact the H_2O_2 and can be made of a noncompatible material, such as bronze, so as to eliminate galling.

Three methods for making connections in a peroxide line are shown in Fig. 11. At the left is a V-type clamp with Teflon gasket manufactured by the George Dahl Co. In the center is a standard ASA large tongue-and-groove flange with a confined Teflon gasket and at the right a Parker Triple Lok flared tube fitting. None of

the three plastics, polyethylene, Teflon, or Kel-F, which have been found satisfactory for gasketing in H_2O_2 service is elastomeric. Therefore, they must all be completely confined to prevent cold flow.

Fig. 12 shows a polyethylene-lined stainless-steel braided hose manufactured by the Jet Specialties Co. Kel-F lined hoses are also satisfactory for H_2O_2 service, but are more expensive. A spray coating of polyethylene on a mild steel test cup has withstood the action of H_2O_2 at room temperature. At a temperature of about 120 F, the polyethylene apparently permits the diffusion of H_2O_2 . When the H_2O_2 reaches the under-surface, it decomposes and forms blisters which destroy the coating. Teflon-coated and Kel-F coated cups are satisfactory for long-time exposure to high strength H_2O_2 at temperatures up to 160 F.

A Princo Densitrol has been used to automatically measure and record the concentration of H_2O_2 solutions. The density of the solution, and hence its concentration, is determined by electromagnetically measuring the position of a plummet submerged in the solution flowing through a sampling chamber. This instrument is compensated for temperature variations.

Four types of chromel-alumel thermocouples with stainless-steel protective tubes have been used. Two of these are of 300-series stainless steel and are suitable for use with liquid H_2O_2 , while two are for use with decomposition gases in a reactor. One unit, made by the Thermo-Electric Co., has a pronged outlet with a connecting plug to facilitate installation. A small couple, made by the Revere Corp., may be screwed into a "Tee" tube fitting. For use with gases at high temperatures and pressures, the thermocouple should be sealed to prevent blowout of the element in case the metal shield is melted. In one, made by Lewis Engineering Co., the sealing is accomplished by crimping the metal tube, while the Revere unit is sealed by means of a packing nut.

A typical reactor for an H_2O_2 propellant system consists of a unit in which H_2O_2 liquid is introduced

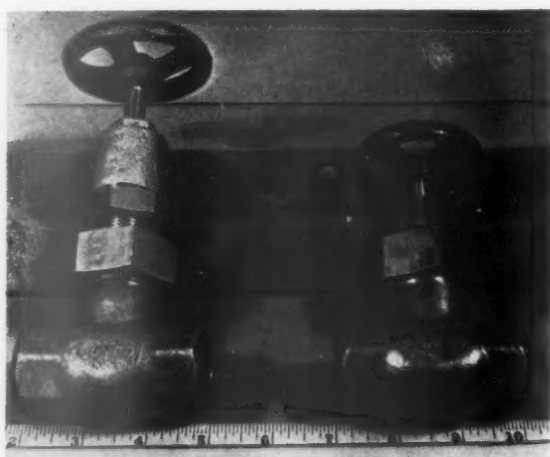


FIG. 10 ALOYCO 1/2-IN. STAINLESS STEEL NEEDLE VALVES

through a nozzle at the top and decomposed by the catalyst to form live steam and oxygen. For a monopropellant system, the decomposition products would be exhausted through a reaction nozzle or led to impinging against a turbine blade. For a bipropellant system, the decomposition products would be used to burn a combustible such as kerosene or ethyl alcohol and the combustion products then exhausted through a jet or led to a turbine.

IV Conclusion

High-strength peroxide can be handled with ease and safety, provided attention is paid to a few simple precautions. Considerable advances have been made in the design of equipment ideally suited for such service. There are available today numerous stock items for the storage, transportation, and use of concentrated hydrogen peroxide.

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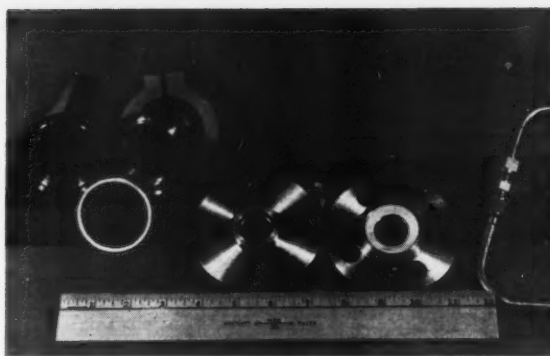


FIG. 11 PIPE AND TUBE CONNECTIONS

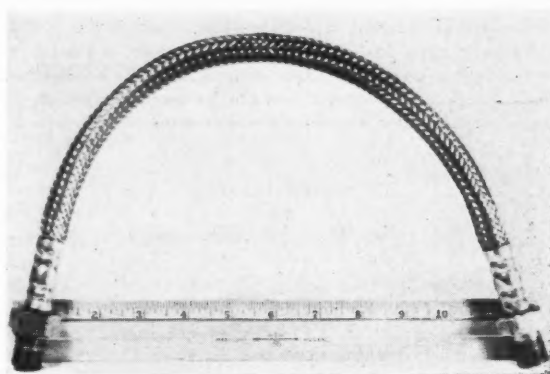


FIG. 12 JET SPECIALTIES $\frac{1}{2}$ -IN. POLYETHYLENE-LINED STAINLESS-STEEL BRAIDED HOSE

Manuscripts Invited for Presentation at the Fall Meeting of the ARS

This meeting is to be held in Chicago, Illinois, September 1-5, 1952, in connection with the Centennial of Engineering, an event sponsored jointly by the principal engineering societies in the United States in commemoration of the founding of the first engineering society one hundred years ago.

Brief abstracts should be submitted as soon as possible to:

Dr. M. J. Zucrow
Program Chairman, ARS
c/o School of Mechanical Engineering
Purdue University
Lafayette, Indiana

Please refer to the inside front cover of the *Journal* for acceptable scope of subject matter. Manuscripts are to be prepared in accordance with style instructions on inside back cover.

Calculation of Adiabatic Decomposition Temperatures of Aqueous Hydrogen Peroxide Solutions

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University of Minnesota, Minneapolis, Minn.

The temperature and states attained on the fractional adiabatic decomposition of aqueous hydrogen peroxide solutions have been calculated using the best available thermodynamic data. The technique of calculation is described and graphical presentations are given of the temperature, and liquid and vapor compositions as a function of the fraction of hydrogen peroxide decomposed for several initial solution concentrations and at several pressures.

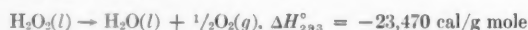
Nomenclature

f	= fraction of initial H_2O_2 decomposed
(g)	= gas phase
h	= enthalpy of superheated steam, Btu per lb
h_f	= enthalpy of saturated water, Btu per lb
H	= enthalpy. H_4 for enthalpy of 1 g mole $\text{H}_2\text{O}_2(g)$ above 20 C, calories; H_5 for enthalpy of $\frac{1}{2}$ g mole O_2 above 20 C, calories
ΔH	= enthalpy change associated with vaporization, decomposition, solution, or pressure change
ΔH_1	= integral heat of solution, calories per g mole $\text{H}_2\text{O}_2(l)$ (see definition in text)
$(H^\circ - H^\circ_g)$	= enthalpy, above 0 K, of hydrogen peroxide in the ideal gaseous state
(l)	= liquid phase
M	= moles H_2O per mole H_2O_2
P	= gas pressure
p_h, p_w	= vapor pressure above pure H_2O_2 and H_2O , respectively, psia
p_{soln}	= vapor pressure above H_2O_2 solution, psia
$p_{\text{H}_2\text{O}}, p_{\text{H}_2\text{O}_2}$	= partial pressure of H_2O and H_2O_2 , respectively, psia
Q	= heat of dilution, calories per g mole H_2O_2 (see definition in text)
R	= gas constant
T	= temperature, degrees Kelvin
t	= temperature, degrees Centigrade
t_f	= adiabatic decomposition temperature, Centigrade
V	= volume of gas per mole
w	= weight per cent H_2O_2 in initial solution
x	= mole fraction of H_2O in H_2O_2 solution
π	= total pressure, lb per sq in., absolute (psia)

THE use of hydrogen peroxide solutions as a source of power, either alone or as an oxygen carrier to support combustion of a fuel, has become possible during the past 15 years by advances in the technology of production to the point that high strength aqueous solu-

tions (containing about 80 weight per cent or more, hydrogen peroxide) could be safely manufactured and handled, and stored with an acceptably low rate of decomposition.

For the reaction



the heat of decomposition is sufficient under adiabatic conditions to vaporize all the water initially present plus that formed by the decomposition, if the initial hydrogen peroxide concentration is greater than about 68 weight per cent. For example, the complete decomposition in a steady-flow process at 515 psia of 83 per cent hydrogen peroxide yields a superheated steam-oxygen mixture at 575 C which is capable of generating slightly over 0.1 hp-hr per lb of original 83 per cent solution if the decomposition products are expanded to one atmosphere through a reversible work-producing device. A more effective use of the products of decomposition is to employ the oxygen to burn a fuel. During World War II hydrogen peroxide solutions of about 80 to 85 per cent were utilized in Germany as a source of power for many weapons. For example, it was used as a monopropellant to drive the fuel pumps of the V-2 rocket, and in the launching units for the V-1 bomb; and as an oxygen carrier for a fuel in rocket-propelled aircraft and the propulsion systems of submarines and torpedoes. These have been described elsewhere (1, 2, 3).³

In most of the uses the hydrogen peroxide is catalytically decomposed—a process which is extremely complex chemically. Starting with a liquid feed at ambient temperatures, decomposition reactions take place in the liquid and/or vapor phases and over a wide temperature range. The physical equilibrium states and temperature attained by hydrogen peroxide solutions undergoing fractional adiabatic decomposition at constant pressure have been calculated to aid in the study of the decomposition mechanisms, and to assist in an understanding of the application of hydrogen peroxide decomposition as a source of power in jet or other types of engines. This paper describes a method for making such calculations and presents some results of interest.

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² Assistant Professor of Chemical Engineering.

³ Numbers in parentheses refer to References on page 77.

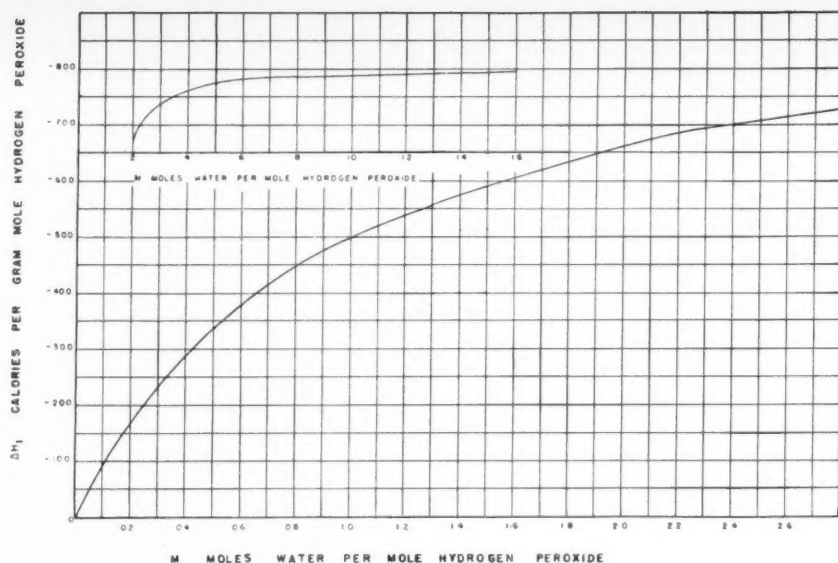


FIG. 1 INTEGRAL HEAT OF SOLUTION

Thermal and Equilibrium Data

The following data are required: (a) Integral heats of solution; (b) heat of decomposition of pure $\text{H}_2\text{O}_2(l)$; (c) vapor-liquid equilibria for the system $\text{H}_2\text{O}_2\text{-H}_2\text{O}$; (d) vapor pressure data for the system $\text{H}_2\text{O}_2\text{-H}_2\text{O}$; (e) enthalpies of $\text{H}_2\text{O}_2(g)$ and l , $\text{O}_2(g)$, and $\text{H}_2\text{O}(g)$ and l as a function of temperature and pressure.

The heat of decomposition and the integral heats of solution were obtained from a set of heat of dilution data recommended by Mickley (4). Mickley's evaluation was based primarily on the unpublished heat of dilution data of H. L. Pike and H. Green, and was made prior to the more recent data of Kubaschewski and Weber (5) on partial molal heats of solution of water in hydrogen peroxide. A comparison of the two sets of data on a comparable basis shows satisfactory agreement in the more concentrated solutions. Mickley's evaluations have been used for the decomposition calculations here and his set of data is given in Table 1.

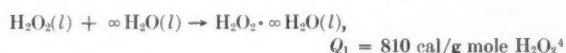
TABLE 1 HEAT OF DILUTION AND HEAT OF DECOMPOSITION OF AQUEOUS HYDROGEN PEROXIDE SOLUTIONS

Weight per cent H_2O_2	Heat of dilution to infinite dilution, Q , cal/g mole H_2O_2	Heat of decomposition, ^a cal/g mole H_2O_2
100	810 ^b	23,470 ^b
90	635	23,295
80	485	23,145
70	360	23,020
60	260	22,920
50	160	22,820
40	80	22,740
30	40	22,700
20	25	22,685
10	15	22,675
0	0	22,660

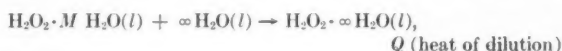
^a Heat evolved by the reaction: $(\text{H}_2\text{O}_2 + M \text{H}_2\text{O})(l) \rightarrow (1 + M)\text{H}_2\text{O}(l) + \frac{1}{2}\text{O}_2(g)$. Data are for 20°C and 1 atmosphere pressure.

^b Extrapolated.

The concepts of heat of dilution and integral heat of solution may be represented by the following symbolism. Heat of dilution of 100 per cent hydrogen peroxide:



For the dilution of a solution containing M moles of water per mole of hydrogen peroxide to infinite dilution,



The difference between these equations represents the integral heat of solution, $(Q - 810)$, in calories per gram mole of hydrogen peroxide.



Values of the integral heat of solution are plotted in Fig. 1.

The thermodynamic properties of gaseous hydrogen peroxide have been estimated by Mickley (4) from available infrared spectrum data. Mickley's tabulations are given in Table 2. $(H^\circ - H_0^\circ)$ represents the

TABLE 2 ENTHALPY OF HYDROGEN PEROXIDE VAPOR

Temp., °K	$H^\circ - H_0^\circ$ RT
291.16	4.078
298.16	4.089
400	4.244
500	4.450
600	4.676
700	4.906
800	5.129
900	5.334
1000	5.547
1100	5.738
1200	5.918

⁴ Note that a positive value of Q means heat is liberated. Thus $Q = -\Delta H$.

enthalpy above 0 K of hydrogen peroxide in the ideal gas state. ($H_{293.16}^\circ - H_0^\circ = 2377$ cal/g mole.) The sensible enthalpy of hydrogen peroxide vapor is presented in Fig. 2.

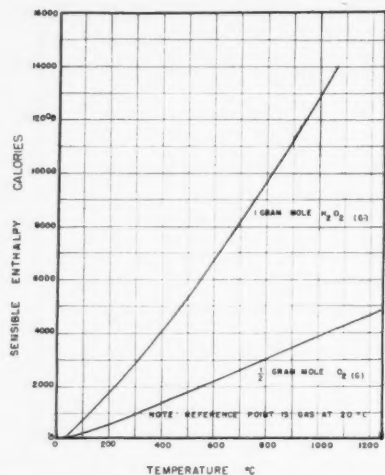


FIG. 2 ENTHALPY OF HYDROGEN PEROXIDE VAPOR AND OXYGEN

Scatchard, Kavanagh, and Ticknor (6) have recently measured the vapor pressure above aqueous hydrogen peroxide solutions as a function of composition and temperature. These data represent a refinement over the earlier work of Giguère and Maass (7). From the vapor pressure measurements alone, as a function of concentration, plus the vapor pressures of pure water and pure hydrogen peroxide, the molal volumes of the two pure liquids, and deviation of the two vapors from the perfect gas law, it is possible to calculate the change with composition of the chemical potential (partial molal free energy) and other thermodynamic properties. [See, for example, Scatchard and Raymond (8).] This method was used by Scatchard, Kavanagh, and

Ticknor to develop thermodynamic equations for the hydrogen peroxide-water system for the temperature range of 0 to 150 C, from which vapor compositions may be computed.

For the present calculations it was convenient to use their equations to construct a Dühring chart, shown in Fig. 3, in which straight line extrapolation to 300 C was made. This chart is a plot of the temperature at which a hydrogen peroxide solution has a given vapor pressure against the temperature at which pure water has the same vapor pressure (solid lines) and also a plot of the temperature at which a hydrogen peroxide solution has a given partial pressure of hydrogen peroxide against the temperature at which pure water has numerically the same vapor pressure (broken lines). As an example of the comparison of the use of the Dühring chart with the thermodynamic equations of the above authors, consider a hydrogen peroxide solution at 258.5 C containing a mole fraction of water equal to 0.49. From the Dühring chart, the partial pressure of hydrogen peroxide is equal to the vapor pressure above pure water at 164.2 C, which is found from the steam tables to be 100 psia. The corresponding value calculated from the equations of Scatchard, Kavanagh, and Ticknor is 99 psia.

The heat of vaporization of pure hydrogen peroxide as determined from the data of Scatchard, Kavanagh, and Ticknor is $\Delta H = 18,411 - 25.815T + 0.021066T^2$. At a temperature of 293.16 K, $\Delta H = 12,650$ cal/g mole H_2O_2 .

Enthalpies for molecular oxygen in the ideal gas state were obtained from Woolley's evaluations (9) based upon recent values of the physical and spectroscopic constants and are shown in Fig. 2.

The Keenan and Keyes Steam Tables were used to obtain the enthalpies for water and steam. For some of the calculations involving decomposition of 100 per

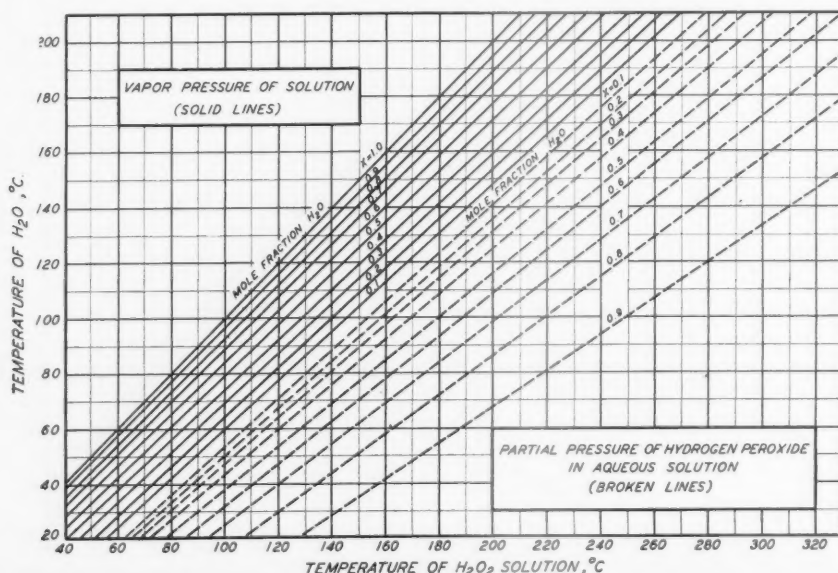


FIG. 3 DÜHRING CHART FOR HYDROGEN PEROXIDE-WATER SOLUTIONS

cent hydrogen peroxide, it was necessary to extrapolate slightly above the maximum temperature of 1600 F for which data are given in these tables.

Methods of Calculation

The calculation of the adiabatic decomposition temperature requires the use of an energy balance in which it is convenient to visualize the over-all process as consisting of the sum of a number of separate physical or chemical processes so chosen that the energy change associated with each may be readily evaluated. The schemes used for the calculations of the adiabatic decomposition temperature as a function of fractional decomposition of hydrogen peroxide, weight per cent hydrogen peroxide in the initial solution, and pressure are illustrated in Fig. 4. The exothermic nature of the decomposition is such that for sufficiently concentrated

Several methods of estimating the effect of pressure on enthalpy of gases are summarized in a separate following section. The applicability of Dalton's Law was assumed.

Single-Phase Region

Basis: 1 gram mole of hydrogen peroxide in M gram moles of water at 20 C. Let f = fraction of hydrogen peroxide decomposed. Assume a trial value of t_f = adiabatic decomposition temperature, C, corresponding to f .

The solution of the energy balance is by successive approximations requiring evaluation of the following quantities:

Step	Enthalpy change, cal.
1 Separate liquid H_2O_2 and H_2O	$-\Delta H_1^a$
2 Decompose f g moles of H_2O_2	$-23,470 f$
3 Vaporize $(1-f)$ g moles of H_2O_2	$12,650 (1-f)$
4 Heat $H_2O_2(g)$ to t_f , °C	$(1-f)H_4^b$
5 Heat $O_2(g)$ to t_f , °C	fH_5^c
6 Vaporize and heat $(M+f)$ g moles of H_2O to t_f , °C	$(M+f)(h-36.04)(18/1.8)^d$

^a ΔH_1 is the integral heat of solution. See Fig. 1.

^b H_4 is the enthalpy of 1 g mole of $H_2O_2(g)$ above 20 C. See Fig. 2.

^c H_5 is the enthalpy of $1/2$ g mole of $O_2(g)$ above 20 C. See Fig. 2.

^d The enthalpy h for superheated steam is found in the steam tables at the state corresponding to the partial pressure of the water and the final temperature. (It is necessary to convert from the steam table base of 32 F to the thermodynamic base used here of 20 C.) The partial pressure of the water is

$$p_{H_2O} = \frac{(\pi)(M+f)}{(1+M+f/2)} \text{ psia,}$$

where π represents the total pressure (psia) at which decomposition occurs.

The assumed value of t_f is correct if the following energy balance is satisfied:

$$23,470f + \Delta H_1 = (12,650)(1-f) + (1-f)H_4 + fH_5 + (M+f)(h-36.04)(18/1.8)$$

Two-Phase Region

Basis: 1 gram mole of hydrogen peroxide in M gram moles of water at 20 C. Let f = fraction of hydrogen peroxide decomposed. Assume a trial value of t_f = adiabatic decomposition temperature, deg C, and $(1-x)$ = mole fraction of hydrogen peroxide in remaining liquid solution, corresponding to f . The solution for this case involves a double set of successive approximations, for it is necessary to assume trial values for both the final temperature, t_f , and the mole fraction of the remaining solution $(1-x)$. A material balance serves as a check on the mole fraction assumed and an energy balance, for the temperature assumed. For any given fractional decomposition, it is assumed that the vapor is in physical equilibrium with the remaining liquid, and that Dalton's Law is valid for the calculation of the amounts of water and hydrogen peroxide evaporated.

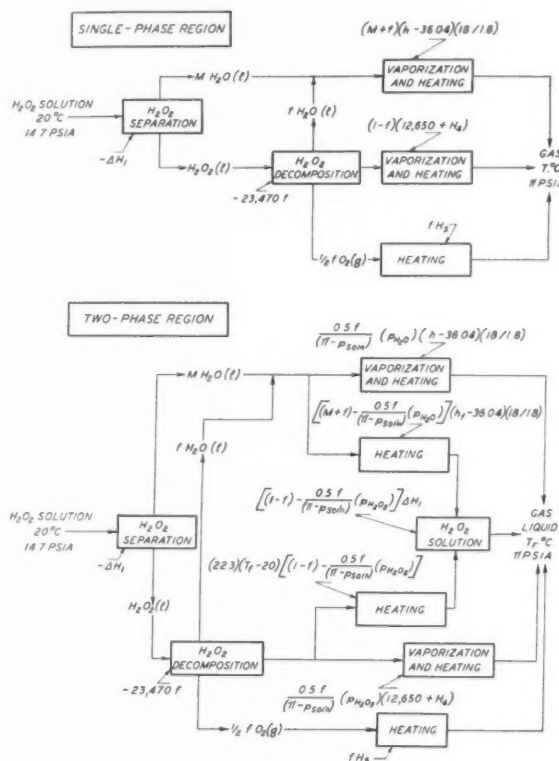


FIG. 4 CALCULATION SCHEMES

solutions all the products of decomposition, including the initial water content, are in the vapor state. This region is termed the Single-Phase Region. For the conditions under which insufficient heat is liberated to vaporize completely the remaining solution, the term Two-Phase Region is applied, and a different scheme of calculation is used. The methods of calculation for the two different schemes are summarized below for the case of a liquid solution initially at 20 C.

Several approximations are involved in the energy balances. Changes in enthalpy with pressure were neglected for the liquids and for the mixing of the gases.

Material and Energy Balances for Two-Phase Region

1. Determine the vapor pressure above the liquid phase, p_{soln} , and the partial pressures of water, $p_{\text{H}_2\text{O}}$, and hydrogen peroxide, $p_{\text{H}_2\text{O}_2}$, from the Dühring plot.

$$2. \text{ Gram moles } \text{H}_2\text{O} \text{ evaporated} = 0.5f [p_{\text{H}_2\text{O}}/(\pi - p_{\text{soln}})]$$

$$\text{Gram moles } \text{H}_2\text{O}_2 \text{ evaporated} = 0.5f [p_{\text{H}_2\text{O}_2}/(\pi - p_{\text{soln}})]$$

$$3. \text{ Gram moles } \text{H}_2\text{O} \text{ remaining in solution} = (M + f) - (\text{g moles } \text{H}_2\text{O} \text{ evap.})$$

$$\text{Gram moles } \text{H}_2\text{O}_2 \text{ remaining in solution} = (1 - f) - (\text{g moles } \text{H}_2\text{O}_2 \text{ evap.})$$

$$4. (1 - x) = \frac{(1 - f) - 0.5f [p_{\text{H}_2\text{O}_2}/(\pi - p_{\text{soln}})]}{(1 + M) - 0.5f [p_{\text{soln}}/(\pi - p_{\text{soln}})]}$$

Calculations are repeated for a fixed trial value of t_f until the selected value of $(1 - x)$ checks.

The steps involved in the energy balance are similar to the previous case, but several additional factors have to be taken into account. Allowance is made for the remixing of the unvaporized water and hydrogen peroxide. The specific heat of the solution is not known at elevated temperatures. As an approximation, the sensible enthalpy change of the remaining solution (heated from 20 C to t_f) is taken to be the sum of the sensible enthalpy change for pure liquid water and for pure liquid hydrogen peroxide. The steam tables were used for the water term and a constant heat capacity was assumed for the peroxide term. More precise evaluations for the heat capacity of liquid hydrogen peroxide solutions (from the temperature dependence of heat of vaporization and sensible enthalpy of hydrogen peroxide vapor) are not warranted. The assumption made, then, is that the integral heat of solution does not have a strong temperature dependence. The double trial-and-error solution is satisfied if the material balance and the energy balance (system enthalpy change = 0) are both satisfied.

TABLE 3 ENTHALPY CHANGE FOR INCREASE IN PRESSURE FROM 14.7 TO 515 PSIA, AT 600 C.

	— ΔH , cal/g mole mixture —		
	100% H_2O	76.5% $\text{H}_2\text{O} + 23.5\% \text{O}_2$	100% O_2
1 Stockmayer method (10)	...	81	...
2 $\Delta H = \int_{14.7}^{515} \left[\frac{\partial(V/T)}{\partial(1/T)} \right]_P dP$ integrated from P - V - T data	116.5	..	-3
3 Generalized enthalpy-difference charts (11)	142
4 Generalized enthalpy-difference charts, using mole fraction combination of critical constants	...	97	...
5 Keenan and Keyes Steam Tables	115
6 Correction for enthalpy of steam only, evaluated from steam tables at the partial pressure of steam (see "Methods of Calculation")	...	65	...

Effect of Pressure

The effect of pressure on the enthalpy of gaseous mixtures of water, hydrogen peroxide, and oxygen has not been measured, but reasonable estimates of the effect can be made. Throughout the major range of the Single-Phase Region, the mole fraction of hydrogen peroxide in the gas phase is small and the effects of pressure and heat of mixing on the enthalpy of hydrogen peroxide vapor have been neglected. A comparison is made in Table 3 above of the results of several methods of calculation of the effect of pressure on the enthalpy of steam, oxygen, and mixtures of the two. Such calculations were made for 100 per cent steam, 100 per cent oxygen, and a mixture of 76.5 per cent steam and 23.5 per cent oxygen (decomposition products of 83 weight per cent solution) for a pressure change from 14.7 to 515 psia, at 600 C. It is seen that the steam contributes the major effect and that the presence of oxygen in the

ENERGY BALANCE FOR TWO-PHASE REGION

Step	Enthalpy change, cal
1 Separate liquid H_2O_2 and H_2O	$-\Delta H_1$
2 Decompose f g moles of H_2O_2	$-23,470f$
3 Vaporize fraction H_2O_2	$(12,650) (0.5f) \left(\frac{p_{\text{H}_2\text{O}_2}}{\pi - p_{\text{soln}}} \right)$
4 Heat $\text{H}_2\text{O}_2(g)$ to t_f , °C	$0.5f \left(\frac{p_{\text{H}_2\text{O}_2}}{\pi - p_{\text{soln}}} \right) H_4$
5 Heat $\text{O}_2(g)$ to t_f , °C	fH_5
6 Vaporize and heat H_2O fraction to t_f , °C, $p_{\text{H}_2\text{O}}$ psia	$0.5f \left(\frac{p_{\text{H}_2\text{O}}}{\pi - p_{\text{soln}}} \right) (h - 36.04) (18/1.8)$
7 Heat remaining $\text{H}_2\text{O}(l)$ to t_f , °C	$\left[(M + f) - 0.5f \left(\frac{p_{\text{H}_2\text{O}}}{\pi - p_{\text{soln}}} \right) \right] (h_f - 36.04) (18/1.8)^a$
8 Heat remaining $\text{H}_2\text{O}_2(l)$ to t_f , °C	$(22.3) (t_f - 20) \left[(1 - f) - (0.5f) \left(\frac{p_{\text{H}_2\text{O}_2}}{\pi - p_{\text{soln}}} \right) \right]^b$
9 Mix remaining liquids	$\left[(1 - f) - (0.5f) \left(\frac{p_{\text{H}_2\text{O}_2}}{\pi - p_{\text{soln}}} \right) \right] \Delta H_1$

^a h_f is the value in the steam tables corresponding to saturated liquid at t_f .

^b The heat capacity of liquid H_2O_2 is taken to be 22.3 cal/(g mole) (°C).

steam mixture reduces the deviation from ideality. With the method employed in this paper of correcting only the steam at its corresponding partial pressure (Method 6), a slightly lower calculated temperature results than would be the case with the use of either of the other two methods.

Since the heat capacity of the steam-oxygen mixture cited above is about 9 cal/g mole of mixture, it is seen that for this mixture a change in the pressure level of decomposition from 14.7 to 515 psia results in a final adiabatic decomposition temperature about 7 C higher, by the steam-table method of calculation used here. The final temperature is about 9 C higher by the Stockmayer method and about 11 C higher by the generalized-chart method. It is thus seen that the calculated results are not significantly affected by the method used to estimate effect of pressure on gas enthalpy.

Results of Calculations

Single-Phase Region

Adiabatic decomposition temperatures at 515 psia for the Single-Phase Region are given in Fig. 5 and Table 4. The range of temperature is from 272 to 1006° C, for initial solution concentrations of from 67.6 to 100 weight per cent hydrogen peroxide. Above

TABLE 4 CALCULATED ADIABATIC DECOMPOSITION TEMPERATURES FOR SINGLE-PHASE REGION

Solution initially at 20 C. Pressure = 515 psia

Fraction H ₂ O ₂ decomposed	Weight fraction H ₂ O ₂ in initial soln—						
	1.0	0.95	0.90	0.85	0.80	0.75	0.70
	Temperatures in °C						
1.00	1006	877	749	624	499	382	272
0.95	919	793	672	...	429	317	
0.90	830	703	590	472	358	257	
0.85	741	...	511	...	290		
0.80	653	539	430	319			
0.75	561	...	344				
0.70	467	326	261				
0.65	372	270					
0.60	272						

TABLE 5 CALCULATED ADIABATIC DECOMPOSITION TEMPERATURES FOR TWO-PHASE REGION

Solution initially at 20 C. Pressure = 515 psia

Fraction H ₂ O ₂ decomposed	Weight fraction H ₂ O ₂ in initial soln—				
	1.0	0.90	0.80	0.70	0.50
	Temperatures in °C				
1.00	225
0.90	239	...
0.80	226
0.70	251	242	...
0.60	...	258	249	...	215
0.50	271	257	...	238	...
0.40	242	...	175
0.30	265	241	...	199	...
0.20	237	197	170	...	102
0.10	157	128	...	84	...

a temperature of 300 C, the following equation represents the results within 10 C.

$$t_f = -1984 + 490f + w(12 + 13f),$$

where

t_f = adiabatic decomposition temperature in °C

f = fraction of hydrogen peroxide decomposed

w = concentration of initial solution, weight per cent hydrogen peroxide

The effect of pressure in the single-phase region is relatively small, as indicated by Fig. 7.

Two-Phase Region

In the Two-Phase Region the range of adiabatic decomposition temperatures is less than that for the

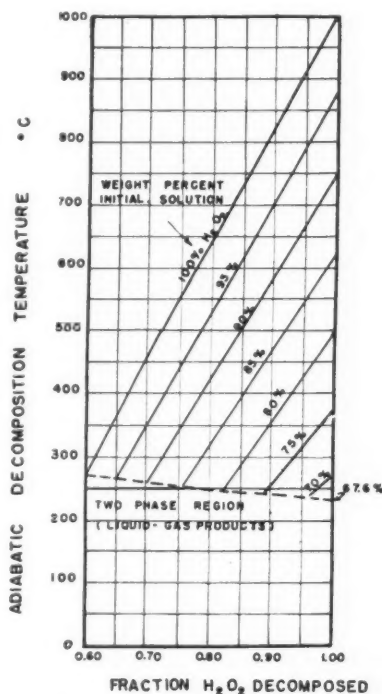


FIG. 5 ADIABATIC DECOMPOSITION TEMPERATURES IN THE SINGLE-PHASE REGION AT A PRESSURE OF 515 PSIA

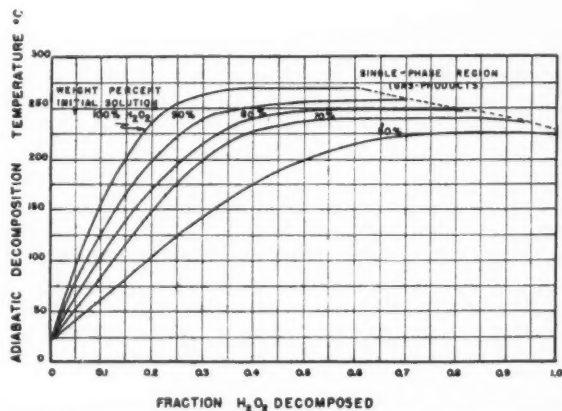


FIG. 6 ADIABATIC DECOMPOSITION TEMPERATURES IN THE TWO-PHASE REGION AT A PRESSURE OF 515 PSIA

Single-Phase Region. For example, with a 70 weight per cent solution over a range of decomposition for $f = 0.5$ to 0.95 , the temperature variation is less than 10°C . Fig. 6 is a plot of the results calculated for a decomposition pressure of 515 psia; corresponding values are given in Table 5. The effects of pressure are illustrated for both 90 and 50 weight per cent solution in Figs. 7 and 8. At the lower pressures, the flatness of the temperature curve is accentuated.

Another interesting aspect of the decomposition is the relatively high hydrogen peroxide concentration in the liquid phase due to the high relative volatility of the water. As shown in Fig. 9 for an initial 90 weight per cent hydrogen peroxide solution (mole fraction 0.8266), the mole fraction in the liquid phase ranges from 0.6 to 0.5 over a range of decomposition from $f = 0.3$ to $f = 0.7$; at 515 psia. Over the same range of decomposition, the variation in the composition of hydrogen

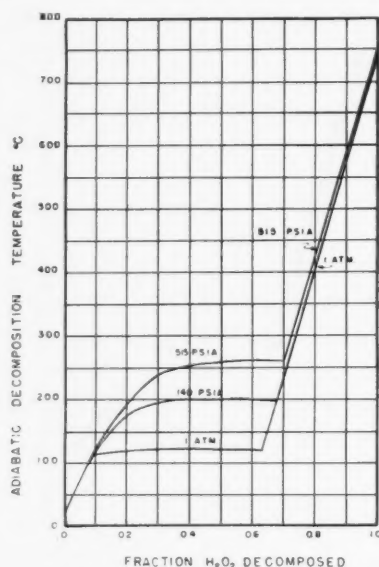


FIG. 7 EFFECT OF PRESSURE ON THE ADIABATIC DECOMPOSITION TEMPERATURE OF A SOLUTION INITIALLY 90 WEIGHT PER CENT H_2O_2

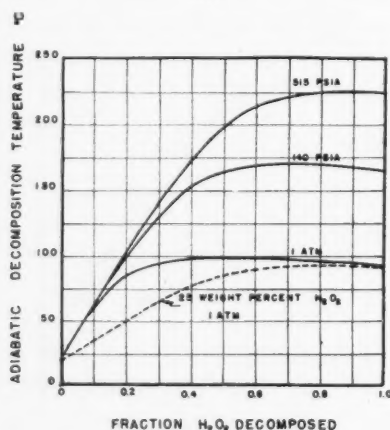


FIG. 8 EFFECT OF PRESSURE ON THE ADIABATIC DECOMPOSITION TEMPERATURE OF A SOLUTION INITIALLY 50 WEIGHT PER CENT H_2O_2

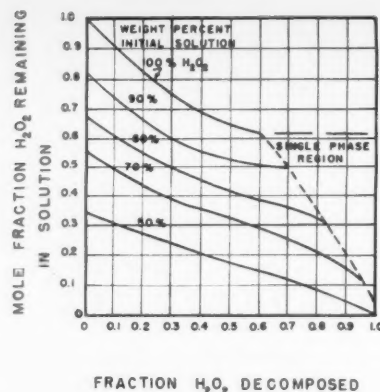


FIG. 9 COMPOSITION VARIATION OF THE LIQUID PHASE ON ADIABATIC DECOMPOSITION AT A PRESSURE OF 515 PSIA

peroxide in the gas phase is less than 0.03 mole fraction. The composition of the gas phase at 515 psia is illustrated in Fig. 10, for a solution initially 90 weight per cent in hydrogen peroxide.

Enthalpy Distribution

The enthalpy contributions of the products of decomposition at 515 psia are graphically represented in Fig. 11 for a solution initially 90 weight per cent hydrogen peroxide. The integral heat of mixing the remaining liquid hydrogen peroxide and water is included inasmuch as the sensible heats of the pure liquids have been computed separately. Representations of this type can be used to estimate the uncertainty in the final temperature for a given error in the primary data used for the calculations. Only for the initial fractional decomposition does the sensible heat of the liquid hydrogen peroxide represent a major term. The decomposition temperature in this region depends strongly on the evaluation of the sensible heat of the remaining liquid, which is probably known with less accuracy than any of the other data used. However, in and approaching the Single-Phase Region, the enthalpy of the water vapor is a major factor and the calculated values become more reliable.

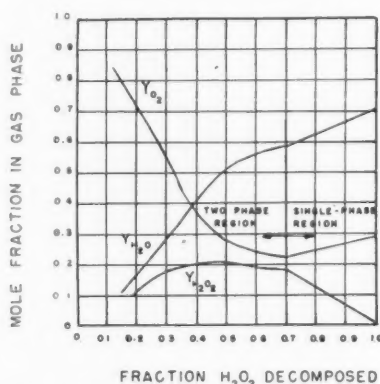


FIG. 10 COMPOSITION VARIATION OF THE GAS PHASE ON ADIABATIC DECOMPOSITION AT A PRESSURE OF 515 PSIA. SOLUTION INITIALLY 90 WEIGHT PER CENT H_2O_2

Change of Initial States

All calculations reported here have been made for peroxide solutions initially at 20 C. For the estimation of enthalpy differences due to changes in the initial temperature of the hydrogen peroxide solutions, the following heat capacities may be used:

Weight per cent H_2O_2	Heat capacity of solution, cal/(g mole of H_2O_2) ($^{\circ}C$)
50	55
60	44
70	36
80	31
90	26.5
100	22.3

NOTES: REFERENCE TEMPERATURE 20 $^{\circ}C$
ENTHALPY OF LIQUID EQUALS
ENTHALPIES OF $H_2O_2(L) + H_2O(L) - \Delta H_M$

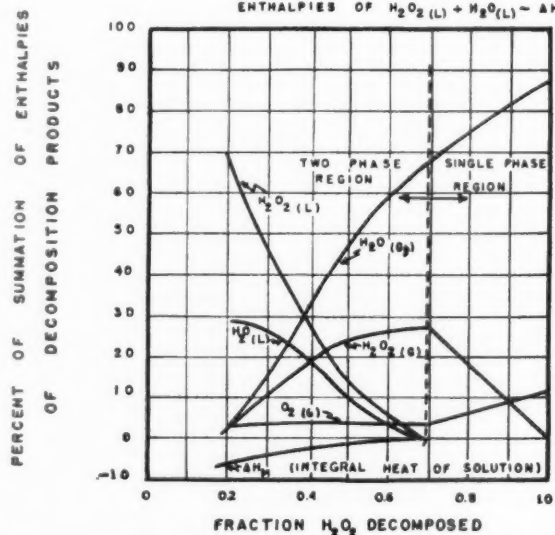


FIG. 11 ENTHALPY DISTRIBUTION OF DECOMPOSITION PRODUCTS FROM A SOLUTION INITIALLY 90 WEIGHT PER CENT H_2O_2 , AT A PRESSURE OF 515 PSIA

Over the major portion of the Single-Phase Region, heat capacities of the vapor range from about 13 to 18 cal/(g mole of initial H_2O_2) ($^{\circ}C$). Using the above values it is seen that changing the initial temperature of an 80 per cent solution 10 C, changes the decomposition temperature about 20 C.

Acknowledgment

The calculations reported here were first performed at the Massachusetts Institute of Technology under Navy Bureau of Ordnance Contract NOrd 9107, Task C. After the introduction of the data of Scatchard, Kavanagh, and Ticknor, the calculations were repeated with the new data by J. McIrvine, under the auspices of the University of Minnesota Graduate School Grant in Aid of Research.

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Combustion Studies with a Rocket Motor Having a Full-Length Observation Window

By KURT BERMAN¹ and STANLEY E. LOGAN²

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A 3-in. chamber diameter, 1200-lb thrust rocket motor with a $1/8$ -in. wide by $8^{21}/32$ -in. long window was built and used for a photographic investigation of the rocket motor combustion process. A 35-mm General Radio continuous strip camera and a Fastax high-speed movie camera were the chief photographic tools used. Pictures are presented illustrating the ignition, preliminary, and full-stage sequence of rocket motor combustion during both stable and unstable operation. The reactants used were ethyl alcohol and liquid oxygen.

Introduction

ENGINEERS have been confronted with many new problems in the course of rocket motor development. These problems include fundamental questions such as the manner in which the combustion process proceeds, and the effect of different methods of fuel and oxidizer injection.³ Various complex unsteady phenomena, generally designated by the terms "combustion instability" or "rough" burning have been encountered at times. Their origin and elimination have been the source of much speculation. Hence, it was decided to "take a look inside" an actual rocket chamber operating under normal conditions. For this purpose a rocket motor with a long observation window was built to permit a photographic investigation of combustion. To date, this motor, which has a total reactant flow rate of 5 to 6 lb per sec and develops about 1200 lb of thrust, has been operated successfully with various injector heads for 47 separate test runs. The fuel-oxidizer combination used in all tests was ethyl-alcohol liquid oxygen. It is the purpose of this paper to present results from four of these runs, representing three different types of injector heads during both stable and unstable operation.

Equipment

The rocket motor body (Fig. 1) designed and built for this study has a 3-in. diam, 10-in. long cylindrical chamber, machined from a Troadaloy 1 (copper-beryl-

lium-cobalt-alloy) forging. A wall thickness of $2\frac{1}{2}$ in. acts as a thermal capacitor, permitting uncooled test runs of about 10 sec duration. The window is supported by a $1/4$ -in. thick shelf at the bottom of a $5/8$ -in. wide window well. A slit $1/4$ in. wide by $8^{21}/32$ in. long extends toward the nozzle from a point even with the injector head face. The principal components of the window assembly are two polished quartz plates, each $1/2$ in. thick, separated by a $1/2$ -in. spacer. Nitrogen gas is admitted to the space between these plates and flows around the inner one into the combustion chamber, thus protecting the quartz from hot gases. Flow to the window space is maintained constant at about 0.1 lb/sec by feeding the nitrogen through a small sonic nozzle from a line pressure regulated to 820 psig. Thus the nitrogen flow into the motor represents roughly 2 per cent of the total flow rate through the chamber.

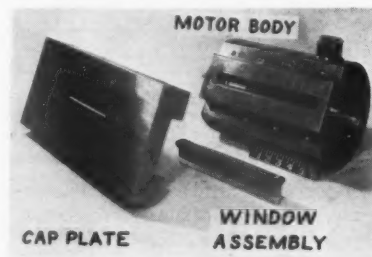


FIG. 1 WINDOW MOTOR CHAMBER DISASSEMBLED

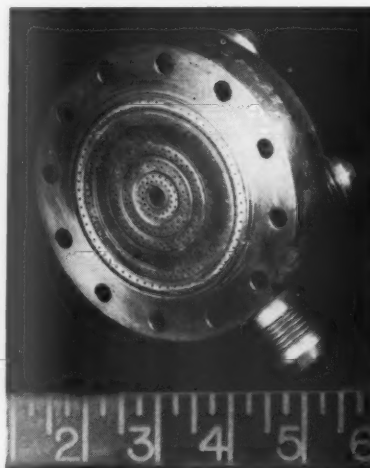


FIG. 2 INJECTOR HEAD NO. 3

Presented before the AMERICAN ROCKET SOCIETY at its Annual Meeting, Atlantic City, N. J., on November 30, 1951.

¹ Research Engineer. Member ARS.

² Research Engineer.

³ See, for example: NACA Report RME8F01, "Photographic Study of Combustion in a Rocket Engine. I. Variation in Combustion of Liquid Oxygen and Gasoline with Seven Methods of Propellant Injection," by D. R. Bellman and J. C. Humphrey, June 1948 (unclassified).

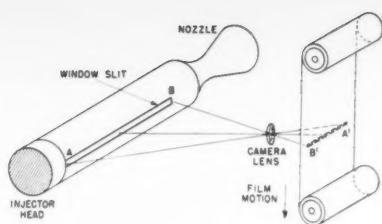


FIG. 3 SCHEMATIC SKETCH OF FILM MOTION RELATIVE TO MOTOR SLIT

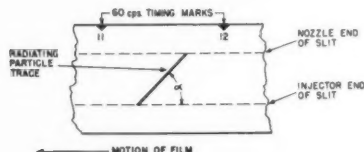


FIG. 4 SKETCH OF FILM TRACE OF PARTICLE MOVING ALONG THE SLIT AT CONSTANT VELOCITY

Recesses are arranged for mounting normal developmental nozzles and injector heads. A nozzle with a 41-deg convergent half-angle, 1.75-in. throat diam, and 15-deg divergent half angle was used for this series of experiments. Each injector head used has an injector plate consisting of a series of concentric rings containing drilled injector holes, alternate rings being for fuel and oxidizer. In a "nonimpinging" head (Fig. 2) the injector holes are drilled parallel to the rocket motor axis, while in an "impinging" head, pairs of liquid jets impinge on each other. Table 1 lists details of the modified impinging, conventional impinging, and non-impinging heads used in tests reported here.

Although a 16-mm Fastax movie camera with speeds up to 8000 frames per sec was used in several tests, the most widely used photographic tool was a General Radio Company Type 651, 35-mm continuous-strip camera with a 2-in. focal length lens, operated at film speeds of 8 to 80 ft/sec. This camera has no shutter, the film being continuously in motion. Thus the rocket motor radiation is recorded without interruption. An externally excited spark gap provided 60 cps timing marks on the film edge. Four types of film were used: 100-ft rolls of 35-mm Eastman Kodak Company Super XX, Daylight Type Kodachrome, and Spectroscopic film types 103-O and I-N. Super XX film was used for the tests reported here. As an aid in correlating events with photographs, the following were recorded on a multichannel oscillograph: (1) chamber pressure, (2) strip camera timing marks, (3) voltage pips obtained from energizing various solenoid valves in the reactant lines.

The strip camera was mounted so that its field of view included the complete slit, and the direction of film motion was perpendicular to the longitudinal rocket motor axis. This is shown schematically in

Fig. 3. To help clarify the interpretation of the film record obtained with this camera, consider, e.g., a radiating particle moving along the slit at constant velocity. It will trace a diagonal streak on the moving film, the slope of which (Fig. 4) is a function of the relative velocities of the radiating particle and film. Knowing the film speed from the distance between timing marks, and the magnification factor from the ratio of slit length to its corresponding image, it is possible to calculate particle velocity along the slit from the slope of the trace measured on the film. Assuming, e.g., that the film speed is 10 ft/sec, the magnification factor is 13.7 and the slope angle is 45 deg, the particle velocity becomes

$$v = v_f M \tan \alpha = 10 \times 13.7 \times 1 = 137 \text{ ft/sec} \dots [1]$$

where

$$\begin{aligned} v &= \text{particle velocity, ft/sec} \\ v_f &= \text{film velocity, ft/sec} \\ M &= \text{magnification factor} \\ \alpha &= \text{slope angle of the trace} \end{aligned}$$

Procedure

Ignition of the reactants was accomplished by means of a pyrotechnic igniter inserted into the motor through the nozzle. The igniter sparks were discharged downstream from a point approximately in the center of the combustion chamber. Propellant tanks were pressurized from regulated gas supplies preset to produce the desired reactant flow rates.

Rocket motor combustion was initiated after closing the igniter switch by opening valves in the following order: (1) Preliminary or first-stage oxygen, (2) first-stage alcohol, (3) full alcohol, (4) full oxygen. First-stage operation limits flow rates during the transition period while chamber pressure builds up.

Time intervals elapsing between actuation of the various valves is a function of the judgment and response of the test pit operator. A representative time sequence was found to be the following:

Igniter	First oxygen	First alcohol	FIRST STAGE	Full alcohol	Full oxygen	FULL STAGE
X	X	X	X	X	X	
0.0	0.2	0.7		2.5	3.0	
Time, sec						

Results

A typical example of a strip film record is shown in Fig. 5a. The light spots at the film edges are 60 cps timing marks. In this case, the numbers 51 and 52 indicate that $51/60$ and $52/60$ sec, respectively, have elapsed since the timing was started. The straight line closest to the timing mark is the image of a reference light mounted on the motor near the nozzle end of the slit, while the two bottom lines are images of an injector-end reference light and a 1-sec timing light.

Previous tests with the modified impinging head No. 1 (Table 1), made with a conventional water-cooled body, had indicated stable operation at chamber pressures higher than about 330 psig, while below this



FIG. 5A IGNITION PERIOD SEQUENCE, TEST NO. 1



FIG. 5B IGNITION PERIOD SEQUENCE, TEST NO. 1



FIG. 5C IGNITION PERIOD SEQUENCE, TEST NO. 1

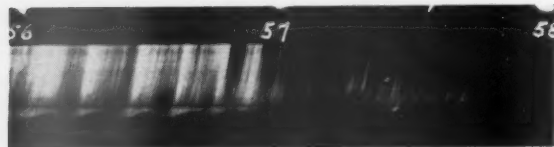


FIG. 5D IGNITION PERIOD SEQUENCE, TEST NO. 1

value there was a tendency to emit a high pitch squeal. A total of 30 window motor tests were made using this head. Two typical tests, one a normal high chamber-pressure run, the other a low chamber-pressure squealing run are discussed here. Beginning this discussion with the high chamber-pressure test (Test 1, Table 2) the ignition period is shown in Figs. 5a-d. The igniter is burning and first-stage oxygen has arrived in the chamber. The first-stage alcohol valve was energized at time mark 31. It should be noted that before time mark 52 there is no radiation visible upstream of the igniter. Halfway between time mark 52 and 53 (Fig. 5b), after alcohol has arrived in the chamber, the radiation strikes back toward the injector. This is followed by a time interval of about $\frac{1}{30}$ sec during which the only radiation visible is downstream of the igniter, except for segments near the injector. The latter vary in brightness at a frequency of about 360 cps. Oscil-

TABLE 1 INJECTOR HEADS

Injector number	1	2	3
Type	Modified impinging	Conventional impinging	Non-impinging
Oxygen			
No. of rings	2	2	3
No. of holes	72	62	258
Pressure drop, psi (at $\dot{w}_0 = 3.1$ lb/sec)	39	17	11
Alcohol			
No. of rings	3	3	3
No. of holes	96	62	258
Pressure drop, psi (at $\dot{w}_f = 2.2$ lb/sec)	20	20	10

TABLE 2 TEST DATA

Test no.	1	2	3	4
Injector no.	1	1	2	3
Film velocity, ft/sec	8	40	10	10
Test duration, sec	7	10	7	10
Chamber press., psig	331	277	330	330
Oxygen tank press., psig	442	364	435	413
Fuel tank press., psig	389	329	383	360
Oxygen flow rate \dot{w}_0 , lb/sec	3.10	2.62	3.08	3.09
Fuel flow rate \dot{w}_f , lb/sec	2.20	1.83	2.15	2.31
Ratio = \dot{w}_0/\dot{w}_f	1.41	1.43	1.43	1.34

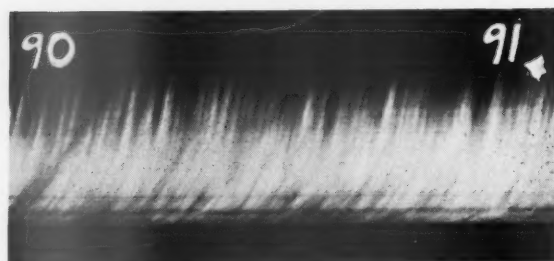


FIG. 6 FIRST-STAGE OPERATION, TEST NO. 1

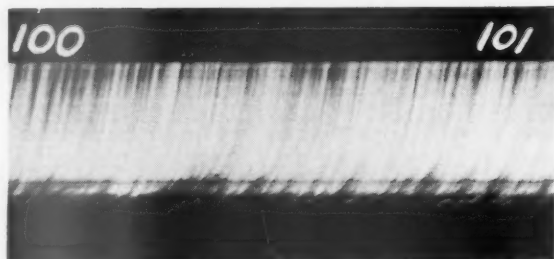


FIG. 7 FIRST-STAGE OPERATION, TEST NO. 1

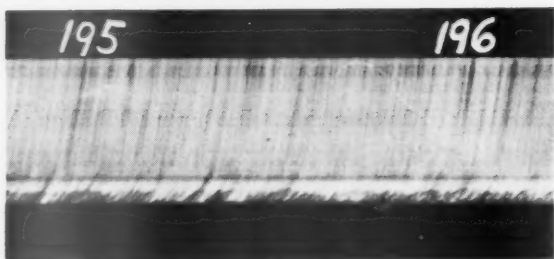


FIG. 8 FULL-STAGE OPERATION, TEST NO. 1

lations continue for about 0.03 sec between time marks 55 and 56 (Fig. 5c) at a slightly decreasing frequency, as full slit-length radiation develops. Between time marks 53 and 54, i.e., about 0.01 sec after the initial flash back, the igniter is injected.

The chamber pressure record indicates the instant of

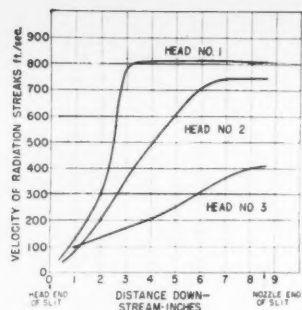


FIG. 9 PROPAGATION VELOCITY OF RADIATION SOURCE ALONG THE SLIT UNDER STABLE OPERATION CONDITIONS (FOR ASSEMBLIES WITH HEADS 1, 2, AND 3)

radiation flash back as a sudden pressure pulse which decays to practically zero. Chamber pressure begins to rise again when radiation develops along the full slit length. Chamber-pressure oscillations between time marks 55 and 57 have a frequency of about 350 cps which agree with the radiation intensity oscillations mentioned previously.

Near the nozzle end, the sources of the radiation have a propagation velocity of approximately 750 ft/sec (streak velocity) as indicated by the slopes of the streaks in Fig. 5d.

Starting at time mark 57 (Fig. 5d) and lasting for a period of roughly $\frac{1}{2}$ sec, the radiation pattern is random. This condition is illustrated at time mark 90 in Fig. 6, where low-velocity streaks of about 200 ft/sec appear superimposed on the high velocity streaks of about 750 ft/sec near the nozzle.

This random pattern grows smoother (Fig. 7) reaching a fully developed first-stage combustion pattern by the time the full-stage alcohol valve is energized. The increase in the alcohol flow rate causes no discernible change in the radiation pattern. Following the energizing of the full-stage oxygen-valve at time 147, full-stage combustion begins (time mark 195, Fig. 8). It is seen that the combustion pattern is nonuniform in time, even for the "smooth" run illustrated here, as

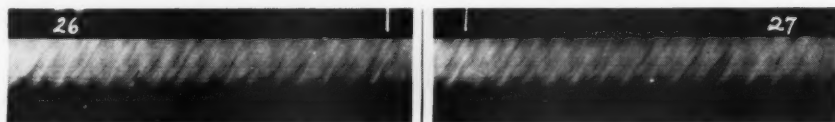


FIG. 10A FULL-STAGE, UNSTABLE OPERATION, TEST NO. 2

evidenced by the basic radiation pattern of closely spaced alternating dark and light streaks. At the injector end, there is a spotty low-velocity region. These streaks extend, within the accuracy of measurement, right up to the injector head. It is found that the average time from a point where they enter the field until they leave is about 0.002 second (stay time). A representative velocity distribution along the slit, plotted in Fig. 9, indicates a maximum velocity of about 800 ft/sec which appears constant beyond a point roughly 3 in. from the injector.

Test No. 2 (Table 2) was a low chamber-pressure

squealing run with the same injector head No. 1. The motor began emitting a high-frequency squeal as soon as full-stage operation had been reached. A typical radiation pattern during this squealing operation is shown in Figs. 10a and 10b. Examining this picture in detail, one notices radiation streaks moving downstream at changing velocities, indicating radiation emitters which undergo acceleration and deceleration along the slit length. Streak velocities are oscillatory both in distance and in time. Fig. 11a is a plot of velocity as a function of distance downstream from the injection end of the slit for an arbitrarily selected radiation streak, while Fig. 11b is a corresponding plot of local streak velocity as a function of time for a fixed point midway down the slit.

Propagating in an upstream direction, at a frequency of 1550–1620 cps, one observes what are believed to be pressure waves reflected from the nozzle end. These become visible as they move upstream because the pressure pulses cause ignition of unburned combustible gases by a process of rapid compression. Correlation with chamber-pressure oscillations is not possible in this case since the pressure gage used was not sensitive to frequencies of the above magnitude. It is likely that pressure waves also travel downstream from the injector toward the nozzle, but pictures obtained to date have not confirmed this clearly. Any such pressure waves tend to be masked by background radiation, as they must travel through a region which is already luminous due to previous ignition.

The slope of the reflected wave path, being rather fuzzy, is difficult to determine accurately. Measurements of about 25 different lines (Fig. 10b) yielded an average absolute velocity of 2170 ft/sec. Adding to this the average downstream gas velocity of approximately 800 ft/sec, the velocity of a reflected wave, relative to the gas, becomes about 3000 ft/sec, a value which differs not too greatly from the sonic velocity (about 3800 ft/sec) corresponding to the estimated chamber temperature. Assuming that pressure waves

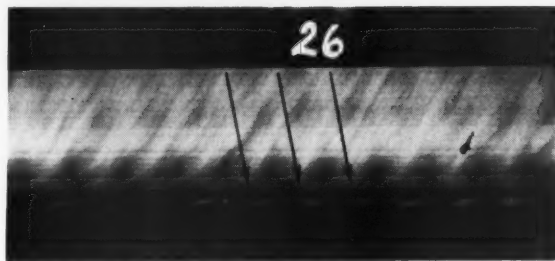


FIG. 10B FULL-STAGE, UNSTABLE OPERATION, TEST NO. 2, REFLECTED PRESSURE WAVES INDICATED BY LINES

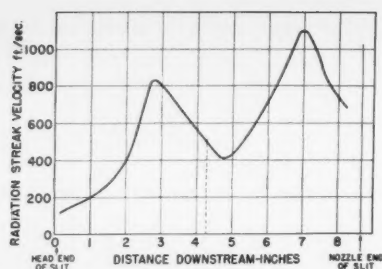


FIG. 11A VELOCITY DISTRIBUTION ALONG THE SLIT FOR AN ARBITRARILY SELECTED SOURCE DURING UNSTABLE TEST NO. 2

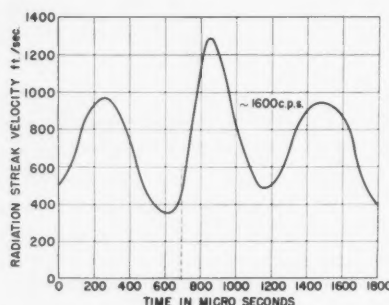


FIG. 11B VELOCITY VARIATION AT A FIXED POINT DURING UNSTABLE TEST NO. 2

move downstream, in this case at an absolute velocity of $3000 + 800 = 3800$ ft/sec, and that reflection occurs at the end of the 10-in. long (0.83 ft) motor body, an "organ-pipe" frequency of this motor can be calculated, based on the assumption that it acts like a closed end tube. Thus the time for a wave to complete its round trip becomes

$$t = \frac{L}{v_1} + \frac{L}{v_2} = \frac{0.83}{2200} + \frac{0.83}{3800} = 0.000597 \text{ sec, which represents a frequency of } 1675 \text{ cycles/sec} \dots [2]$$

where

- t = period, sec
- L = length, ft
- v_1 = upstream velocity, ft/sec
- v_2 = downstream velocity, ft/sec

a value which agrees well with the observed pressure reflection frequency of 1560–1620 cps.

As an example of a conventional impinging type injector head test, a typical run with injector No. 2 (Table 1) is illustrated by test No. 3 (Table 2). This injector head produces a characteristic noisy start, i.e., the motor howls during the starting sequence.

The pattern of the ignition period is similar to that exhibited by head No. 1. Shortly before the full-alcohol valve was actuated, there occurred a short oscillation period, lasting about $1/6$ sec, having a frequency of approximately 1400 cycles/sec. After the full-stage alcohol valve was energized (time mark 98) a low-frequency oscillatory interval began, as illustrated in Fig. 12 (time mark 104). These oscillations continued during the first oxygen-full alcohol period, with the frequency slowly decreasing from about 340 cps at time mark 104 to a steady value of about 240 cps (time mark

147, Fig. 13). Pressure oscillations of corresponding frequencies appeared on the chamber pressure record. The full oxygen valve was energized at time mark 148. Near time mark 161 (Fig. 14) the oscillations began to decay with a resulting smooth full-stage combustion pattern (Fig. 15) except for one short period of $1/6$ sec when low-frequency oscillations suddenly appeared again.

It may be seen that the full-stage radiation pattern has an appearance different from that obtained with head No. 1. A $1/2$ -in. section at the injector end exhibits very little radiation structure. Beyond this section, the streaks attain a velocity of about 750 ft/sec. The stay time is approximately 0.003 sec.

Test No. 4 (Table 2) represents a typical test with the nonimpinging type injector head No. 3 (Table 1). The ignition period was essentially the same with this head as with the previously discussed heads. However, the first stage appears to be slightly smoother, as shown in Fig. 16. Full-alcohol and full-oxygen valves were energized at time marks 156 and 184, respectively. At time mark 205 (Fig. 17) the first indication of an oscillation occurs, the frequency at this point being about 360 cps. Not only are the periods of oscillations intermittent, but the frequency itself changes continuously until it finally reaches a value of 220 cps after several seconds of full-stage operation. An example of a rather smooth pattern is illustrated in Fig. 18. It may be seen (1) that there is a section about 1 in. long at the injector end in which there is little radiation structure visible, and (2) that the streaks do not attain a constant velocity before leaving the field of view. The velocity at the nozzle end of the slit is about 400 ft/sec, and the stay time 0.004 sec.

Analysis

A cautious interpretation of the results presented here is necessitated by the realization that the radiation appearing in the pictures is a projection of a three-dimensional phenomenon. The film records only the radiation which emanates from the surface of the flame volume. It must also be kept in mind that the 0.01 lb/sec of nitrogen introduced into the motor around the inner window might modify the usual combustion process. At the present time no data are available to evaluate the seriousness of the complications resulting from these factors. However, the radiation intensity oscillations on the strip camera film correspond to chamber pressure oscillations on the pressure record. Thus, it is assumed that the conditions recorded in the photographs are representative of what happens simultaneously across the complete cross section of the flame volume.

The basic radiation pattern of closely spaced streaks, which exists during both stable and unstable operation, does not seem to correspond to any one frequency nor to any repetitive frequency spectrum, as indicated by the irregularity of the streak spacing and brightness. When considering the interpretation of the streaks, one

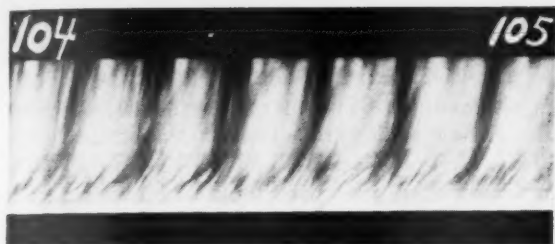


FIG. 12 LOW-FREQUENCY OSCILLATORY PERIOD DURING FIRST OXYGEN-FULL ALCOHOL OPERATION, TEST NO. 3

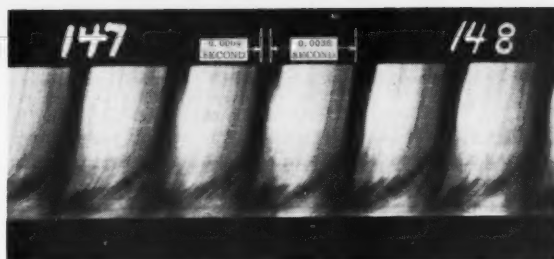


FIG. 13 LOW-FREQUENCY OSCILLATORY PERIOD DURING FIRST OXYGEN-FULL ALCOHOL OPERATION, TEST NO. 3



FIG. 14A DECAY OF LOW-FREQUENCY OSCILLATION UPON ARRIVAL OF FULL-STAGE OXYGEN TEST NO. 3



FIG. 14B DECAY OF LOW-FREQUENCY OSCILLATION UPON ARRIVAL OF FULL-STAGE OXYGEN TEST NO. 3

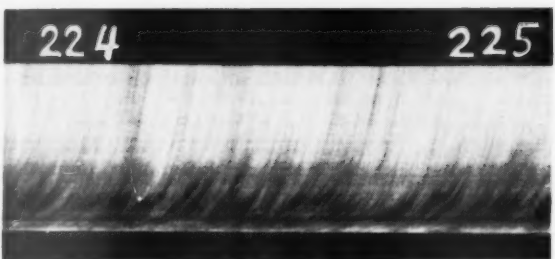


FIG. 15 FULL-STAGE OPERATION, TEST NO. 3

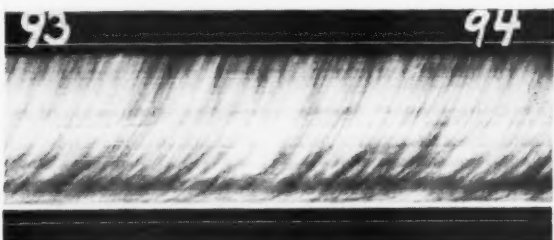


FIG. 16 FIRST-STAGE OPERATION, TEST NO. 4

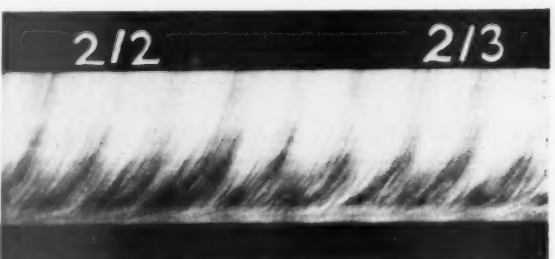


FIG. 17 FULL-STAGE, UNSTABLE OPERATION, TEST NO. 4

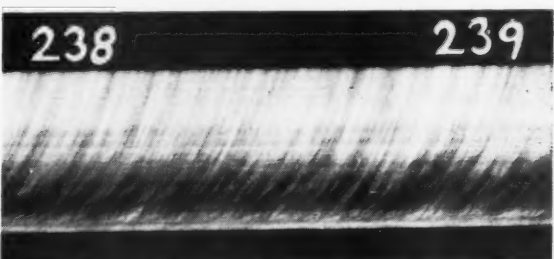


FIG. 18 FULL-STAGE, STABLE OPERATION, TEST NO. 4

can think of at least three factors which can cause fluctuations in the intensity of gas radiation; namely, the composition of the gas mixture, the pressure, and the temperature. Since these are interrelated, it is difficult to apportion the cause of the fluctuations among them. Hence, it is assumed that the bright streaks on the photographic positives are evidence of high-pressure, high-temperature gas, while the darker streaks represent lower pressure, cooler gas regions.

Careful study of the original film seems to indicate that the 200-ft/sec low-velocity streaks, previously pointed out in Fig. 6, are radiation attenuators. These dark streaks may be random masses of reactants or cool gases passing downstream between the window and the bulk of the combustion gases.

Typical velocity distributions during smooth operation, plotted in Fig. 9 for the three types of injector heads, indicate considerable difference in the rates of completion of combustion. The pictures do not indicate a sharp flame front; but the regions in which the velocity gradients level off may indicate the respective depths of maximum chemical reaction rate zones. Although the velocity gradient of the nonimpinging head No. 3 begins to decrease, the velocity does not become constant within the field of view.

Assuming that the terminal velocity of 800 ft/sec mentioned for head No. 1 is an average velocity across the cross section of the motor chamber, and knowing the cross-sectional area of the 3-in. ID motor and the reactant flow rates, it is possible to calculate the density

and temperature of the gas. The average density of the gas becomes

$$\rho = \frac{\dot{w}}{ua} = \frac{5.50}{800 \times 0.049} = 0.135 \text{ lb/ft}^3 \dots \dots [3]$$

where

- ρ = density of the gas, lb/ft³
- \dot{w} = total fuel and oxidizer flow rate, lb/sec
- u = average velocity of the gas, ft/sec
- a = cross-sectional area of the chamber, ft²

Knowing the chamber pressure to be 331 psig, and assuming a perfect gas law relationship and a molecular weight of 22, one obtains an average temperature of

$$T = \frac{P_c}{(R/M)_p} = \frac{345.6 \times 144}{(1544 \times 0.135)/22} = 5250 \text{ R} \dots \dots [4]$$

where

- T = absolute gas temperature, R
- P_c = absolute chamber pressure, lb/ft²
- R = universal gas constant, ft-lb/F lb-mol
- M = average molecular weight of gas

a value which is fairly close to the theoretical combustion temperature.

It is suggested that the low-frequency (220–360 cps) oscillation patterns observed in these tests are determined by characteristics of the injection system and the liquid lines, but this hypothesis must be examined by means of further tests.

Good agreement was obtained between the observed oscillation frequency for the squealing operation with head No. 1 and a calculated organ-pipe frequency. If tests with bodies of various lengths continue to show this degree of correlation, it may be assumed that calculations of oscillation frequencies based on acoustical closed end tube theory are valid, although the method of the excitation and reflection may be very different from that of an organ pipe.

We would like to make a conjecture regarding the cause and nature of this high-frequency type of instability. If we proceed with the fundamental assumption that the low-velocity gas region existing between the liquid fuel and oxidizer jets, in the vicinity of the injector holes, is the flame-stabilizing or holding zone of a rocket motor, then it is plausible to postulate that the effective separation distance between the fuel and oxidizer stream surfaces is a critical flame holding dimension in which the liquid jets assume a role equivalent to the solid boundary of a tube. In direct analogy with the burning phenomenon encountered for flame propagation in tubes, it is to be expected that there exists a critical minimum or quenching distance at a given pressure and mixture ratio at which stable flame propagation becomes impossible. Since the quenching distance varies approximately inversely as the gas pressure, it follows that lower pressure would make the flame-holding abilities of the zone more marginal, and that below a certain chamber pressure (as in test No. 2) a stable combustion pattern would be impossible for a given injector configuration and fuel-oxidizer combination and ratio. The occurrence of this condition would then become the cause of the high-frequency in-

stability. Ignition, instead of proceeding in a manner predicted by the usual steady-state normal flame-velocity theory, becomes unsteady and is initiated by the impulsive compression of the combustible gases by pressure waves traveling in the motor. An unsteady combustion cycle is visualized as being initiated by the rapid ignition of a mass of combustible mixture near the injector head. This discrete mass of hot gas moves downstream preceded by a pressure wave, part of which is then reflected by the nozzle wall or boundary layer. Moving upstream again, the reflected pressure wave compresses combustible gas it meets along its path sufficiently to cause ignition. Such ignition may continuously maintain or increase the energy of the pressure wave as it propagates upstream. Finally, coming back to the low-velocity region near the injector, the compressive impulse of the wave ignites the accumulated gas mixture. This replenishes the pressure pulse and the cycle repeats itself.

Summary and Conclusions

The window-slit rocket motor has proved to be a useful tool for the study of rocket combustion. Its construction is such that the full length of the combustion chamber may be observed while the rocket is operated under normal conditions. The tests reported here lead to the following conclusions:

1 The time-distance radiation pattern of a rocket motor during stable operation consists of a series of streaks of varying intensity. These appear to indicate the propagation of discrete radiation sources, existing due to random variations of pressure, temperature, and gas composition. The fluctuations do not indicate any one frequency or repetitive frequency spectrum.

2 The propagation velocity distribution, as indicated by slopes of the streaks, is a function of the type of injector head used. In each case, there is a region near the injector end of the motor where the radiation sources undergo rapid acceleration.

The following summary represents stable full-stage operation for the three injectors tested: (a) Modified impinging head (No. 1): The accelerating region starts near the head. A constant velocity of about 800 ft/sec is approached at a distance of about 3 in. from the injector. (b) Conventional impinging head (No. 2): The radiation exhibits no structure for a distance of 1/2 in. from the injector. A constant velocity of about 750 ft/sec is approached at a distance of about 5 in. from the head. (c) Nonimpinging head (No. 3): A section about 1 in. long at the injector end exhibits little radiation structure. Downstream of this region a constant velocity is not reached within the field of view. The velocity at the nozzle end of the slit is about 400 ft/sec.

The regions of high acceleration probably correspond to the zones of maximum chemical reaction rates. The stay times for the three heads were found to be approximately 0.002, 0.003, and 0.004 sec, respectively. A gas temperature of 5250 R near nozzle end was calculated, using measured velocity of 800 ft/sec for head No. 1.

3 Two types of instability were observed:

(a) The low-frequency type of instability (220-360 cps) is distinguished by abrupt and severe variations in radiation intensity. As measured at the nozzle end, a typical example of a cycle during 240 cps oscillations consists of a 3.6-millisecond period during which high radiation intensity persists, followed by a period of 0.4 millisecond during which no radiation is visible on the film. The frequencies determined photographically agree with chamber-pressure oscillation frequencies. It is postulated that this phenomenon is a characteristic of the injection system and the liquid lines.

(b) The high-frequency type of instability (1400-1700 cps) represents an unsteady combustion phenomenon distinguished by pressure reflections, and acceleration and deceleration of the sources of radiation as they move downstream. The velocity of the pressure waves was found to be close to the sonic velocity of the chamber gas. A theoretical discussion for this type of instability is presented.

Experiments reported in this paper represent the first exploratory work with this window-slit rocket motor. It is hoped that further understanding of rocket combustion and its associated phenomena will result from future tests.

Acknowledgments

This work was performed as part of Project Hermes. We wish to thank Army Ordnance for permission to publish the results. In addition we acknowledge the assistance and advice of numerous individuals at the Malta Test Station of Project Hermes which made these experiments feasible. Special thanks are due E. H. Hull, G. Mullaney, and Dr. G. E. Moore.

DISCUSSION

By RICHARD J. PRIEM,¹ and MARCUS F. HEIDMANN¹
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Flight Propulsion Laboratory, Cleveland, Ohio

Dr. Berman and Mr. Logan are to be congratulated

¹ Aeronautical Research Scientist.

Presented at the 1951 Annual Meeting of the AMERICAN ROCKET SOCIETY, Atlantic City, N. J., November 30, 1951.

on the excellent work they have done in applying this unique method of photography to the combustion zone in a rocket engine. Their technique appears to be a promising method for giving particle velocity vs. time in a rocket engine having high-frequency instabilities.

The authors have made a good conjecture regarding the cause and nature of the high-frequency instabilities. Their discussion appears to be complicated, since they have discussed simultaneously two theories on the addition of energy to the wave, where either could fulfill the requirements alone. The first theory explains energy addition on the basis of flame stabilizing characteristics, which would provide the additional energy necessary to sustain the wave. The second theory postulated was that the wave compresses the combustible gases it meets along its path sufficiently to cause ignition. The ignition would increase the energy released and thus provide the energy required to sustain the wave. A high-frequency instability is possible if either of the theories postulated is correct.

The theory on ignition by the wave front is interesting in the light of the data presented. One consideration that must be taken into account with the second theory is the existence of a pressure node somewhere near the center of the chamber. This means that any combustible gas in this region will not be compressed by the wave since it is cancelled by a reverse wave. Hence, no energy addition can be obtained at the node point. The over-all effect of the node point would be that less energy is added to a wave when combustion occurs at a node point. This is a possible explanation of why Head Nos. 2 and 3 did not exhibit high-frequency instabilities while No. 1 did. Fig. 9 shows that injector No. 1 produced a constant gas velocity of 810 ft/sec at 3.5 in. downstream. This indicates that the combustion occurs in the front third of the chamber. Injectors Nos. 2 and 3 had combustion occurring for over half the chamber length which indicates that the energy available to the wave was considerably less than for injector No. 1, and, therefore, the wave was eventually damped out. For Head No. 1 the additional energy received by the wave caused by the concentration of combustion close to the injector was

(Continued on page 103)

NOTICE TO CONTRIBUTORS

Manuscripts submitted for publication *must* be double-spaced originals. There should be wide margins at both sides of the sheet and triple spacing around formulas to allow for the marking of directions for the printer.

Original drawings (jet black India ink on white paper or tracing cloth) should accompany the manuscript. (*Blueprints are not acceptable.*) Photographs must be on glossy white paper. The *smallest* lettering on 8- by 10-in. figures should be no less than 1/4 in. high.

Only the simplest formulas should be typewritten, all others should be carefully written in pen and ink. The difference between capital and lower-case letters and Greek symbols should be clearly distinguished.

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Photographic Techniques Applied to Combustion Studies— Two-Dimensional Transparent Thrust Chamber¹

By JOHN H. ALTSEIMER²

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A two-dimensional transparent thrust chamber, approximating a slice, 0.47 in. thick from a 1000-lb thrust chamber, was used for tests of five different injector types. Motion pictures at 2000 and 3000 frames per second were taken of the combustion process. The different combustion patterns and some examples of combustion instability are illustrated. Curves are shown for chamber gas velocities versus axial distances from the injector faces, and the velocities obtained with each injector type are compared.

Introduction

INCREASINGLY, rocket development personnel are recognizing the need for apparatus which will measure or indicate the phenomena occurring within combustion chambers. Several years ago Bellman and Humphrey of the NACA (1)³ ably demonstrated the use of a two-dimensional, transparent chamber for photographing the combustion flame. Recently, the Aerojet Engineering Corporation has conducted tests with similar apparatus which further substantiates the usefulness of such a chamber. The photographic results obtained, while only of a preliminary and exploratory nature, indicate that the technique is worthy of much more attention than it has received.

Five different injection patterns were investigated, using a liquid bipropellant system.⁴ Information was desired on the appearances of flame fronts, striations, recirculations, starting characteristics, and propellant injection streams and droplets. Results were to be compared with information previously obtained on the operation of the injectors in a 1000-lb-thrust, three-dimensional chamber.

Apparatus

The "two-dimensional" chamber resembles an axial slice, 0.47 in. thick, cut from a 1000-lb thrust chamber having a chamber-to-throat area ratio of 5.9. The overall length of the chamber from injector face to the nozzle

exit is approximately 16 in., and the height in the combustion-chamber section is 4.375 in. The flat sides are transparent and consist of two Lucite plates per side: an inner plate $\frac{1}{2}$ in. thick and an outer plate $\frac{3}{4}$ in. thick. The inner plates were replaced after each test, although the outer plates were used repeatedly. A view of the chamber assembly mounted on the test stand is shown in Fig. 1. The seal between the Lucite sides and the metal parts was made with O-ring material. The chamber characteristic length is 57 in., and the design chamber pressure is 300 psia.

The injection patterns simulated, as nearly as possible, those used in 1000-lb thrust chambers. The five patterns tested were as follows:

1 The "showerhead," repetitive-pattern has seven fuel and six oxidizer orifices, as shown in Fig. 2. The streams are nonimpinging, being injected parallel to the longitudinal axis of the chamber, and all lie in one plane. The "repetitive-pattern" designation is from three-dimensional injector nomenclature and refers to a uniform array of holes equally spaced across the injector face. The object is to approach, as closely as is practical, the condition of an infinite number of holes producing a uniform and thoroughly mixed outflow of propellants across all areas of the injector face.

2 The 1:1 impinging, nonrepetitive pattern for a 1000-lb thrust chamber consists of a circle of 12 pairs of 1:1 impinging streams with the oxidizer streams outermost. This pattern is called "unique" to distinguish

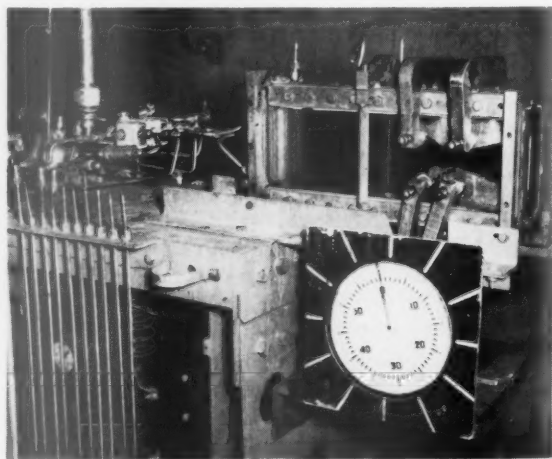


FIG. 1 OBLIQUE REAR VIEW OF TWO-DIMENSIONAL TRANSPARENT CHAMBER ON TEST STAND

¹ Presented at the Annual Meeting of the AMERICAN ROCKET SOCIETY, Atlantic City, N. J., November 30, 1951.

² Development Engineer, Liquid Rocket Engine Department.

³ The number in parentheses refers to the Reference on page 103.

⁴ EDITOR'S NOTE: The author was not permitted by military security officers to disclose the chemical nature of the propellant used in this research. Nevertheless, the editors accepted the paper for publication in the belief that even without this vital information, the results are generally applicable to any propellant system.

it from the repetitive patterns. The "two-dimensional injector," illustrated in Fig. 3, consists of two pairs of these 1:1 impinging streams.

3 The 2:1 impinging, repetitive pattern has eight sets of impinging streams, each set consisting of a center fuel jet being impinged upon at 45-deg. angles by two oxidizer jets. Fig. 4 illustrates this injector.

4 The coaxial-flow, repetitive pattern has eleven sets of coaxial pairs of holes. Each set consists of a center jet of oxidizer surrounded by fuel. Fig. 5 illustrates this injector.

5 The fifth pattern was the splash-wall, repetitive pattern shown in Fig. 6. Seven splash walls (or baffle plates) are used; an oxidizer jet strikes one side of each wall and a fuel jet strikes the other side. The liquids are deflected downstream and meet at the downstream edge of the walls. The splash-wall injector for a 1000-lb. thrust chamber has a grid arrangement similar in appearance to a waffle iron. In the center of each square are four orifices of either oxidizer or fuel which inject streams out to each of the four surrounding walls. The squares alternately enclose oxidizer and fuel orifices.

The cameras used for these tests were Eastman Type III, 16-mm motion picture cameras capable of speeds up to 3000 frames per sec. The film acceleration period used up approximately one half of the film. Actual film speeds were obtained by noting the movement of a timing disk shown on each frame or by using timing pips recorded on the film itself. The total time required to expose the available 100 ft of film was approximately $2\frac{1}{2}$ sec. Kodak Super XX black and white and Kodachrome color film were used.

The $\frac{1}{2}$ -in. thickness of the inner Lucite plate was sufficient to contain the erosion which occurred within the $2\frac{1}{2}$ -sec operating time. However, shortly after the start of a test, the throat area began to enlarge, causing a decrease in chamber pressure. For example, two tests showed decreases of 23 and 41 psi per sec, respectively. The rate of decrease varied according to the particular operating conditions and is an inherent disadvantage of the chamber design. Plates from some typical tests were measured to determine the amount of Lucite lost in the chamber section between the injector and the beginning of the convergent nozzle section. This loss amounted to from 1 to 7 per cent of the propellant flow.

The fuel used was somewhat objectionable because it deposited an opaque substance on the wall in spots where the fuel density was great. Also, liquids flowing along the walls can cause confusion. Both difficulties must be minimized by careful observations.

Measurements of chamber and injection pressures were made with Baldwin SR-4 strain-gage-type pressure pickups and also with Aerojet reluctance-type pickups. The data were recorded on a Consolidated Engineering Corporation Model 5-114-P4 oscillograph. Also recorded on this instrument were the film pipping voltage, timing marks for the camera and the propellant valve operation, and the output from a photolec-

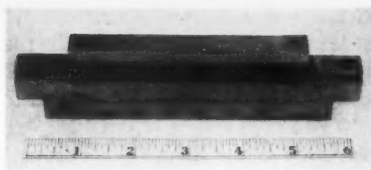


FIG. 2 SHOWERHEAD INJECTOR FOR TRANSPARENT THRUST CHAMBER

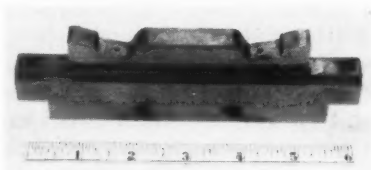


FIG. 3 UNIQUE-TYPE, 1:1 IMPINGEMENT INJECTOR FOR TRANSPARENT CHAMBER

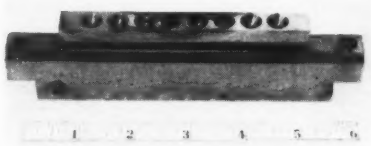


FIG. 4 2:1 IMPINGEMENT, REPETITIVE-PATTERN INJECTOR FOR TRANSPARENT CHAMBER

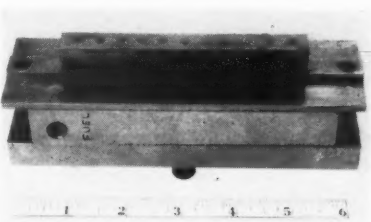


FIG. 5 COAXIAL-FLOW INJECTOR FOR TRANSPARENT THRUST CHAMBER

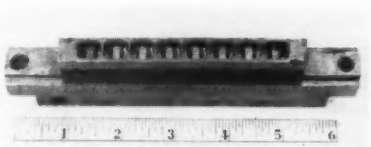


FIG. 6 SPLASH-WALL INJECTOR FOR TRANSPARENT THRUST CHAMBER

tric cell. This cell, used during some of the tests to indicate flame intensity levels, but not absolute intensity magnitudes, permitted correlation of intensity levels with pressures.

Test Results

General Observations

A characteristic common to the repetitive-pattern injectors was the longitudinal flame striations in the combustion patterns. That is, a flame front normal to the chamber axis was not indicated, but rather,

there are columns of flame which correspond with the injection pattern and persist throughout the combustion chamber and throat. With the repetitive-pattern injectors, no large-scale swirls or turbulence patterns were found.

The appearance of the Lucite sides provides confirmation of these observations regarding flame patterns. The sides exhibit erosion streaks which correspond to the hot or cool areas. As was also reported by Bellman and Humphrey (1), the Lucite exit-cone section exhibited Mach line patterns, and the exhaust gases became invisible downstream of the throat. The plates show, also, that the 0.47-in. thickness of the chamber was sufficient to permit striations to exist across this dimension, since the two Lucite plates used in a test showed slightly different erosion patterns. Ideally, the pattern should be the same on both plates, but manufacturing imperfections of the injection orifices prevented this. Examples of the Lucite erosion pattern and the corresponding combustion-flame photograph may be seen in Figs. 7 and 8.

The combustion pattern obtained with the "showerhead" injector is illustrated in Fig. 9. Longitudinal striations were very persistent. Small turbulence eddies were observed near the injector face.

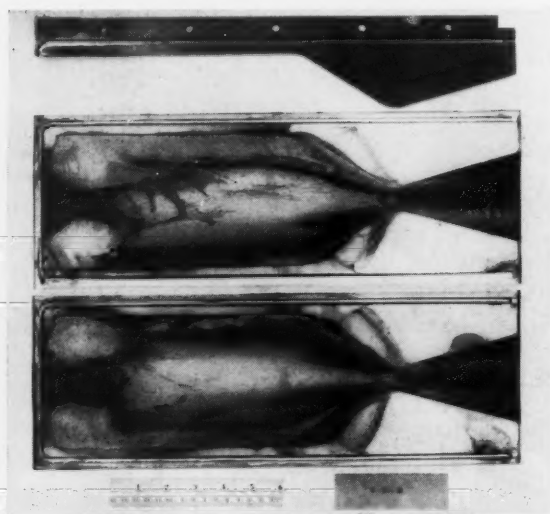


FIG. 7 TYPICAL LUCITE PLATE EROSION PATTERN FOR UNIQUE-TYPE INJECTOR

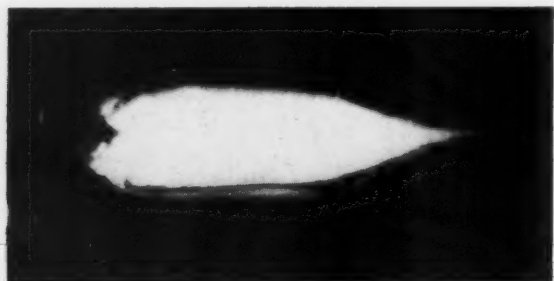


FIG. 8 TYPICAL FLAME APPEARANCE USING THE UNIQUE-TYPE INJECTOR

The combustion pattern obtained with the "unique" injector (Fig. 8) was somewhat different from those obtained with the repetitive-pattern injectors, in that the combustion products exhibited more transverse turbulence. The products swirled to the chamber center line and then moved out through the throat. The brightest part of the flame was on the center line, and dark areas at the top and bottom appeared to be more or less stagnant and essentially unused in the combustion process.

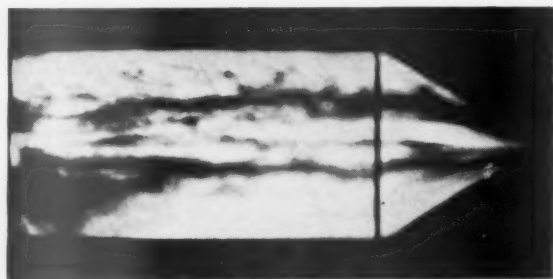


FIG. 9 TYPICAL FLAME APPEARANCE USING THE SHOWERHEAD INJECTOR

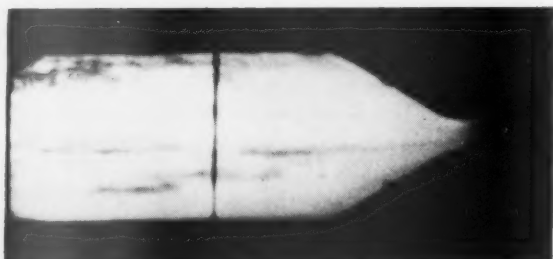


FIG. 10 TYPICAL FLAME APPEARANCE OF THE 2:1 IMPINGEMENT, REPETITIVE-PATTERN INJECTOR

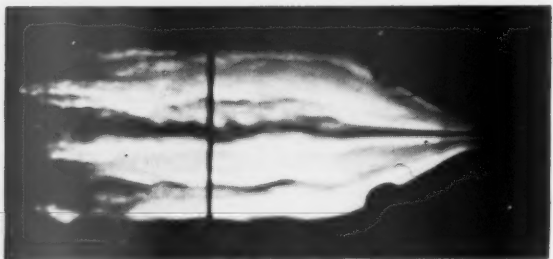


FIG. 11 TYPICAL FLAME APPEARANCE USING THE COAXIAL-FLOW INJECTOR

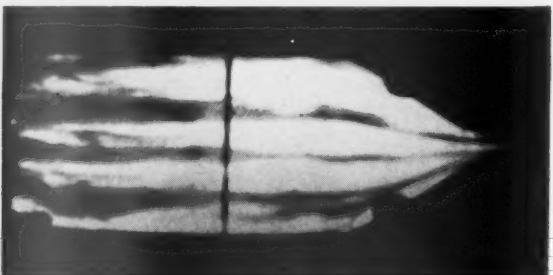


FIG. 12 TYPICAL FLAME APPEARANCE USING THE SPLASH-WALL INJECTOR

With the 2:1 impinging injector the chamber was very well filled with flame (Fig. 10), and it appeared that the ideal repetitive-pattern state was being approached, i.e., uniform conditions throughout the chamber across any plane normal to the chamber axis. However, some longitudinal striations were present with this injector as with the others. Small eddies were noted against the injector face between the set of orifices.

The combustion obtained with the coaxial-flow injector seemed to be utilizing only the downstream portion of the chamber, and the flame pattern was very irregular near the injector face (Fig. 11). Striations existed and considerable flickering between the flame columns was noted.

Strong striations were present with the splash-wall injector (Fig. 12). The dark longitudinal areas correspond to the fuel injection locations, and the bright areas correspond to the oxidizer injection locations. This is a good illustration of the persistence of the injection pattern throughout the chamber.

Combustion Instability

Four of the "unique"-injector tests were characterized by smooth starts followed by "chugging." These four tests were the only ones, out of 13 using this injector, which exhibited strong instability cycles. The four chugging tests were consecutive. The following test was considerably smoother, perhaps because of a plumbing change wherein a piece of $\frac{1}{2}$ -in. tubing, several inches long, was taken out of the fuel line and replaced with $\frac{1}{4}$ -in. tubing of equivalent line volume. Later, in another test with the unique injector, the $\frac{1}{2}$ -in. tubing was reinstalled in place of the $\frac{1}{4}$ -in. tubing, but the chugging did not recur. The $\frac{1}{2}$ -in. tubing was then replaced with a $\frac{1}{2}$ -in. tee, with one leg of the tee capped and acting as a small gas pocket in the line, but again no chugging occurred. No further attempts to determine the cause of the chugging were made because of lack of time and funds for this particular program.

The high-speed pictures of the chugging cycles were examined to determine some of the flame growth characteristics. Two tests were observed, frame by frame, and the places in the thrust chamber where the flame disappeared or was of minimum size and also the manner in which the flame grew to a maximum size again were noted. These observations showed that the flame persists longest on the center line and generally disappears in the throat. Occasionally, the flame does persist near the injector face as well as near the throat. As it redevelops, it starts up in the downstream half of the chamber and spreads toward the injector and the top and bottom walls. Fig. 13 illustrates a typical chugging cycle.

To determine whether the flame-intensity fluctuations were in time phase with the chamber-pressure fluctuations, a photelectric cell was aimed at a point directly under a chamber-pressure pickup, and the cell output was recorded on the oscillograph. The cell was not calibrated, the output being used to indicate the

frequencies and time positions of the maximum and minimum points only. It was found that the flame-intensity fluctuations were almost exactly in phase with the pressure fluctuations.

Chamber Velocities

Eight motion picture films were selected for use in velocity determinations which were representative of the five injectors tested. It was found that many combustion eddies were persistent and identifiable for a long enough period to be recorded on the high-speed camera film. The pictures were projected on a screen, frame by frame, and the distances traveled by the eddies were measured. The actual film speed was determined from the timing-disk data. Knowing the time required to travel the measured distances, the velocities could be calculated. Typical data are shown in Fig. 14 for the "showerhead" pattern. Although the velocities obtained cannot be considered as reliable,

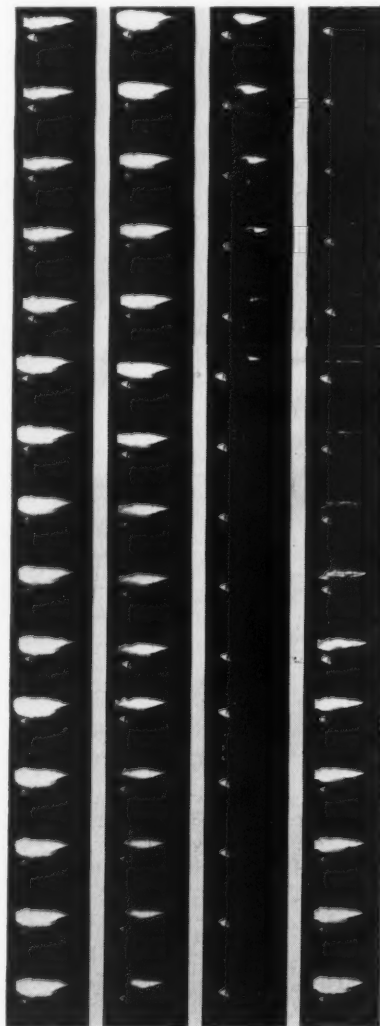


FIG. 13 TYPICAL "CHUGGING" CYCLE OBTAINED WITH THE UNIQUE-TYPE INJECTOR. THE FILM SPEED IS APPROXIMATELY 3000 FRAMES/SEC

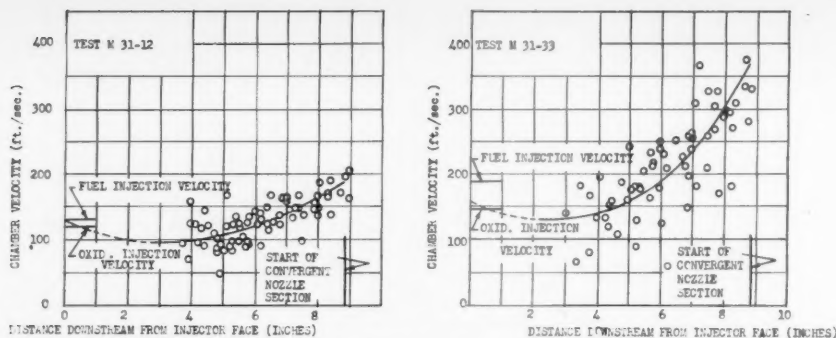


FIG. 14 CHAMBER VELOCITY VERSUS DISTANCE DOWNSTREAM FROM INJECTOR FACE IN TRANSPARENT TWO-DIMENSIONAL THRUST CHAMBER—SHOWERHEAD INJECTOR—VELOCITY MEASURED ON HIGH-SPEED MOTION PICTURES BY TRACKING COMBUSTION EDDIES

the trends of the velocity curves are well established.

A summary of the five most representative curves of velocity versus distance from injector is shown in Fig. 15. No data were obtained for velocities within $2\frac{1}{2}$ or 3 in. of the injector face. The resultant propellant injection velocities were plotted for a distance equal to zero, however, and the curves were arbitrarily extended on dotted lines to meet these points. The limitations in accuracy for the curves presented should be considered and are discussed below:

1 The method of tracking eddies is not a completely positive one. Near the injector the eddies, or dark-spot movements, seen on the film could have been made by liquids; while at the downstream end of the chamber, the eddies were probably due to gases. Within about $2\frac{1}{2}$ in. of the injector, it was impossible to track any eddies, probably because of the high droplet concentration, which was not conducive to the production of distinguishable eddies.

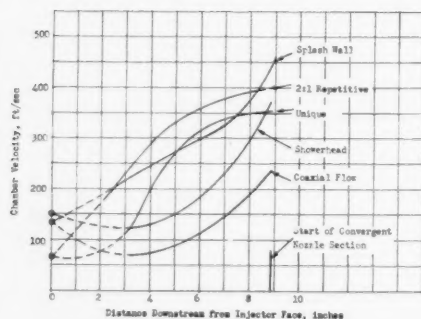


FIG. 15 CHAMBER VELOCITY VERSUS DISTANCE DOWNSTREAM FROM INJECTOR FACE IN TRANSPARENT TWO-DIMENSIONAL THRUST CHAMBER

$L^* = 57$ in. Velocity was measured on high-speed motion pictures by tracking combustion eddies. Camera speeds: 2000 to 3000 frames/sec

Test no.	Injector
M31-24	Unique pattern
M31-29	2:1 repetitive
M31-33	Showerhead
M31-38	Coaxial flow
M31-40	Splash wall

Distance zero velocity equals propellant injection velocity.

2 Two phases of the materials in the chamber are present, the liquid and the gaseous. The liquids enter the chamber at the injection velocity and then immediately begin to slow down. At the same time, the liquids evaporate and react with each other to form gases, whose volumes increase with distance from the injector. So, if the measurement technique is such as to pick up the droplet velocities where the droplet concentrations are high, and the gas velocities where the droplet concentrations are low or nil, then the velocity curves will show an initial downward trend to a minimum value followed by increasing values. The curves, as drawn, take such a shape, especially those curves obtained from the "showerhead" and coaxial injectors. The exact locations or concentrations of the droplets cannot be ascertained from these data. The fact that the curve shapes are as expected is encouraging.

The validity of the curves can be checked at their downstream ends. If the operating data included reliable values of characteristic velocity (C^*), then the combustion temperature could be estimated and the perfect gas law and continuity equations used to determine the velocity. However, since the throat eroded in a variable manner during each test, reliable C^* values are not available. Therefore, a check on the curves was made by using the maximum velocities compared with calculated C^* . These values were used as a basis for judging the reasonableness of the data. The value of chamber pressure (P_c) would, of course, also vary because of throat erosion. However, P_c is a measurable quantity throughout the test. The value recorded on the data sheets and the one used for these C^* calculations is the maximum measured value. Also, to avoid an extreme effect caused by the variable throat area, the velocity data were taken during approximately the first half of each test. If density (ρ), weight flow rate (w_p), and cross-sectional flow area (A_c) are known, then the velocity (v_c) is obtainable from the continuity equation:

$$\rho v_c A_c = w_p \dots \dots \dots [1]$$

Two other equations can be utilized:

$$\rho = \frac{P_c}{RT_c} \dots \dots \dots [2]$$

and

$$\frac{C^*}{C_{\text{theor.}}^*} = \left(\frac{T_c}{T_{c \text{ theor.}}} \right)^{1/2} \dots \dots \dots [3]$$

where

R = chamber gas constant
 T_c = chamber temperature

Combining Equations, [1], [2], and [3] yields

$$C^* = \left(\frac{v_e P_c A_c (C_{\text{theor.}}^*)^2}{w_p R T_{c \text{ theor.}}} \right)^{1/2} \dots \dots \dots [4]$$

Using Equation [4] and substituting the theoretical values of the propellant combination, relative C^* values were obtained. Comparison of these calculated performances with those already known for the 1000-lb-scale injectors showed the performance data to agree within $\pm 20\%$, which is believed to be a reasonable deviation for this case.

The trends of the velocity curves are in accordance with those of curves obtained previously for the various injection designs. The 2:1 repetitive and the splash-wall injectors, which are considered to be high-performance types, show rapid velocity increases after injections; while the nonimpinging types, such as the "showerhead" and the coaxial-flow injectors, require considerably greater downstream travel before a velocity increase is shown. It is encouraging that relatively consistent data were obtained, and it is believed that better velocity data might be obtained with improvements in nozzle constriction. The primary recommendation for improvement would be the use of a metallic, nonerosible throat section and an improved system for tracing the gas movement.

Conclusions

The "two-dimensional" transparent chamber has been found to offer an effective means of studying rocket injection and combustion problems. The tests reported have been useful in comparing the different injector types and the technique is considered to be worthy of further attention. Among the interesting possibilities that suggest themselves are the following:

1 The use of special combinations of light filters and films to locate particular spectral bands and thus assist in analysis of the chemical kinetics involved.

2 Improvement of the tracer technique of measuring velocities in the chamber to obtain further data on the gas-generation characteristics of different injection types.

3 The conducting of similar combustion studies on the principal injector configurations with various propellants selected to represent the several degrees of chemical complexity of rocket propellants and a wide range of physical characteristics.

4 The use of cameras capable of taking pictures at much higher speeds in order to study high-frequency combustion instabilities.

Acknowledgments

The author wishes to thank the various members of

(Continued on page 103)

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Interplanetary Travel Between Satellite Orbits¹

By LYMAN SPITZER, JR.²

Princeton University Observatory, Princeton, N. J.

An analysis is given of the performance to be expected of a rocket powered by nuclear energy, and utilizing an electrically accelerated ion beam to achieve a gas ejection velocity of 100 km/sec without the use of very high temperatures in the propellant gases. While such a rocket would have much too low a thrust to take off from the surface of a planet, it would appear to be capable of traveling from a circular orbit about the Earth to a circular orbit about any other planet in the solar system. Gases obtained from planetary atmospheres could be used for the propellant, and the only refueling required from the Earth would be supplies for the crew and small amounts of fissionable material. Preliminary indications are that such a rocket could feasibly be constructed and operated at the present time.

Introduction

DURING the past decade much thought has been given to the problem of sending a rocket up to a circular orbit some three hundred miles above the earth, and thus creating an artificial satellite. It appears that this problem can be solved by present techniques. With the use of chemical fuels a multi-stage rocket designed to launch a man-made satellite now seems perfectly possible in principle, although much engineering development would be needed to realize this possibility in practice. The ratio of take-off weight to the weight of the satellite would be large, probably several hundred or so, but not prohibitive.

The next step beyond an artificial satellite has seemed much more difficult. To travel from a close circular orbit around the Earth to the Moon and back or to another planet and back requires almost prohibitive amounts of fuel with conventional rockets. In theory, many Earth-launched rockets could be used, each one carrying fuel to one of several refueling stations for an eventual interplanetary trip. However, the many analyses made of this technique indicate that a discouragingly large number of very large rockets would be required to make possible even one trip to the Moon and back. For a planetary trip the difficulties are even greater.

While the possible use of nuclear energy may revolutionize our concepts of space travel, the use of nuclear power simply to heat up the propellant gases from a rocket would not make interplanetary

travel particularly feasible. It is well known that one of the chief limitations on a conventional rocket is the temperature which the rocket tubes can tolerate without melting or evaporating. A nuclear-powered rocket using heated propellant gases could not use a higher gas temperature than a chemically powered rocket, and its only advantage would lie in the use of lighter propellant gases, with higher velocities at a particular temperature. A nuclear-powered rocket making use of molecular hydrogen as propellant would need somewhat less propellant than would a conventional rocket using the best chemical fuels, but the gain would fall far short of the orders of magnitudes of improvement needed to make interplanetary travel feasible. Moreover, the rapid transfer of enormous quantities of heat from a uranium pile to a propellant gas offers difficulties that might well prove insuperable. We may conclude, as others have already done, that the use of nuclear power in an otherwise conventional rocket does not seem to provide an ideal solution to the problem of interplanetary flight, though it might be of use in the trip from the Earth's surface to a close satellite.

This entire picture changes if we shift our point of view and consider space ships which need not land on planets but instead travel from one circular orbit to another. To take off from the Earth and reach a satellite orbit requires not only an enormous amount of energy, but also requires that the energy be delivered in a very short time. A rocket designed to take off from the Earth's surface must obviously have a thrust greater than the rocket's weight, and this requires that the take-off period be short; i.e., the power, or energy per unit time, must be very great. A conventional chemical rocket is the only practical means yet found to produce such enormous powers without excessive weight.

For an interplanetary space ship, however, a large thrust is not required. Such a ship, traveling between circular orbits about different planets, can accelerate relatively slowly, and while the total amount of energy required is still large, the power needed may be reduced by about $1/100$, and instead of the 100,000 hp used in a large chemical rocket, a few thousand hp is sufficient. The present paper describes a low-thrust, low-power ship of this type, designed to travel from one planet to another, without ever landing. The several thousand hp needed is generated in a uranium or plutonium pile, converted into electrical energy, and used to accelerate a stream of ions by purely electrical means to a speed of about 100 km per sec.

¹ A paper read before the Second International Congress on Astronautics in London, September 3-8, 1951.

² Chairman, Department of Astronomy.

EDITOR'S NOTE: The AMERICAN ROCKET SOCIETY was one of the sponsors of the Second International Congress on Astronautics. The current interest of the ARS in space flight has prompted the editors to reprint this article by arrangement with the British Interplanetary Society.

Different components of this ship are discussed in the following sections, with the necessary mathematical analysis given in the appendixes.

The ideas outlined here are probably not new; possibly the extensive literature on space flight contains some reference to a system similar to that proposed in the present paper. No search of the literature has been made.³ The chief purpose of this paper is not to claim priority for any ideas but to focus attention on what promises to be the most practical means for interplanetary flight in the near future.

Power Supply

The power-weight ratio needed to give a ship an acceleration depends only on the acceleration itself multiplied by the velocity of the ejected propellant gas. As a reasonable acceleration we may take 0.3 cm/sec^2 , which would make it possible to escape from a close circular orbit about the Earth in a few weeks and to reach Mars a few months later. The velocity of the ejected gas may be taken as 100 km/sec . A higher value would require more power, while a lower value would yield a less favorable mass ratio. With this choice of acceleration and gas velocity, the power needed is about $1/10 \text{ hp per lb}$, as shown in Appendix 1. We shall consider a ship whose gross weight is about 10 tons, since this is probably the smallest-size ship that could carry a uranium or plutonium pile. The total useful power must therefore be roughly 2000 hp. If an over-all efficiency of one-third is assumed in the conversion of heat into useful power, a heat source of 6000 hp must be provided.

How can this power be produced? A normal pile weighs many hundreds of tons and could presumably generate much larger powers. For an interplanetary space ship one may envisage a pile constructed with U 235, the lighter isotope of uranium, or with plutonium, Pu 239. These materials are the ingredients of an atomic bomb, and a pile made with these fissionable materials could have a much smaller size and weight than one using ordinary uranium. The use of atomic fission for power production is still in the development stage, but one may assume that within the next ten years the design and operation of a small pile, weighing about a ton and producing some 6000 hp of heat, will become entirely practical. Such a pile would consume about 2 kg of U 235 or Pu 239 in a year of steady operation, and nuclear fuel for 50 years of continuous operation would weigh only 100 kg, or a tenth of a ton!

What about the dangerous effects of so much radio-

³ The author is much indebted to Dr. L. R. Shepherd for calling attention to some earlier work along the present lines. Oberth in his book, "Wege zur Raumschiffahrt," proposes a rocket propelled by an ion beam, powered by solar energy. The possible performance of an ionic rocket powered by uranium or plutonium has been considered by Shepherd and Cleaver (*J. Brit. Interplanetary Soc.*, vol. 8, p. 59, 1949); their pessimistic conclusions regarding such a device result from their assumed acceleration of 0.01 gravity, some 30 times the acceleration assumed in this paper.

activity on the crew aboard this ship? The usual pile is surrounded by concrete and lead, to shield personnel from the neutrons and gamma rays. Such shielding is very heavy and would be practical only in an interplanetary ship whose size and weight were already very large. Thus, one could design and probably build a space ship weighing 10,000 tons, powered with a uranium pile weighing perhaps 1000 tons, and generating 2,000,000 hp of useful energy. Such a ship could carry thousands of people and vast supplies anywhere in the solar system, and could even navigate to other stars, though many generations would be born, grow up and die on shipboard before such a journey were complete. However, launching such a ship from the Earth's surface to a close circular orbit would be a tremendous undertaking. With the use of chemical fuels, such a launching would require a rocket of some million tons gross weight, an achievement that would seem far, far in the future.

Returning, then, to our smaller interplanetary ship, we still have the problem of how the personnel are to escape the hazards of nuclear radiation. The easiest way to achieve safety from neutrons and gamma rays is simply by distance. One may envisage the main interplanetary ship as a pilotless engine room, with a light control car, containing the crew, attached by wires some 100 km in length. Power and communications could be sent over the connecting wires, with probably also radio communication and an auxiliary power supply in the control car. Appendix 2 shows that at 100 km from a pile generating 6000 hp of nuclear energy, the flux of neutrons and gamma rays would be down to a safe value. If a shorter towing distance were desired, the pile might be designed as a long thin cylinder, with the axis pointed at the control car, and some shielding provided at the end of the cylinder nearest the control car.

If the car containing the crew and control equipment weighed 2 tons, the wires pulling the car with an acceleration of $0.3 \text{ cm per sec per sec}$ would be subject to a force of 1.2 lb. With a safety factor of 100, the two connecting wires need each have a diameter of only 0.02 in. and would each weigh about 300 lb, not an excessive contribution to the mass of the ship. Careful precautions would be needed to maintain tension on the wires at all times to avert sudden acceleration and breaking of the wires. There seems no reason why this technique could not be used, although so great a distance between passengers and power plant may seem somewhat unconventional. If repairs to the main ship became necessary during flight, the power could be turned off and the control car could be drifted up to the space ship.

Generation of Electrical Power

A uranium or plutonium pile generates power in the form of heat, and this must be converted into electrical energy. One may visualize a small pile, with heavy water used as a moderator, in which the

heavy water is heated up, turned to steam, and used to drive a steam turbine. This turbine then drives a d-c generator, whose output is used to accelerate the propellant gases, as discussed in the next section.

Since no problems of shielding of personnel are involved in the space ship, the design of such a system is somewhat simplified, but many problems of detail remain. The neutrons will produce nuclear transformations all through the ship, with possibly unfavorable results on the operation of the electrical and other equipment. The gamma rays will produce ionization in all materials, again creating problems of equipment performance and maintenance. Considerable research on such problems is presumably under way in connection with present piles, and one may hope that solutions to these problems will be available shortly. There appears to be no reason in principle why such problems cannot be solved.

If the radiation hazard to the ship itself can be overcome, two other problems remain. First, there is the matter of cooling. To operate a heat engine, the heat produced at a high temperature must be given up in a condenser at a low temperature, and the only way for the condenser in a space ship to lose this heat is by radiation. The rate at which a solid body radiates heat varies as the fourth power of the temperature. Thus we find ourselves in something of a dilemma. If the condenser temperature is made low, about room temperature, the rate of radiation is so low that impractically large radiating surfaces are required. If the condenser temperature is made very high, then the temperature at which the heat is generated must be even higher, and materials tend to melt or lose their strength, especially when submitted to nuclear bombardment. This subject is analyzed in Appendix 3, where the temperatures of heat input and heat output are taken to be 900 K and 450 K, respectively, or 1160 F and 350 F. With these temperatures a heat engine has an ideal efficiency of one-half, and at a pressure of about 10 atm the steam will liquefy in the condenser. Even at this relatively high temperature, however, the radiating surface required is about a thousand square yards. One may visualize a thin fin some 30 yd square extending on one side of the ship, radiating the heat produced by the pile.

Secondly, there is the problem of the weight of all the material required to produce some 2000 hp of useful electric power. An over-all weight-power ratio 10-lb per hp is in the same general neighborhood as the weight-power ratio for Diesel-electric railroad locomotives and for bombing planes. It is not unreasonable to assume that about this same value can be achieved for an interplanetary space ship.

Propulsion of Ship

It now remains to convert the electrical power, whose generation was discussed in the preceding section, into useful work. In particular, the only way a

space ship can be propelled is by ejection of a stream of gases, and the electrical power must be used for this purpose. By use of electrostatic forces to accelerate a beam of ions, very high gas velocities can be achieved without the use of very high temperatures. The production of intense ion currents has been extensively studied in the past decade, and the acceleration of a space ship by an ion beam seems to offer no particular difficulties.

The electrical voltage to be applied depends only on the mass of the ion to be used and the velocity desired. The equations, given in Appendix 4, show that to accelerate nitrogen ions to a speed of 100 km/sec requires a potential of 730 volts. Nitrogen is indicated as a propellant, since this gas is readily obtained from the Earth's atmosphere, and the ship can therefore obtain propellant gases in its circular orbit; this avoids the necessity of bringing tons of propellant up from the Earth's surface for every trip. Since nitrogen is probably also abundant in the atmospheres of Mars and Venus and possibly other planets, propellant gases could also be obtained at various points in the solar system.

At the relatively low voltage of 740 volts required to accelerate nitrogen ions, space-charge effects limit the total ion current that can be accelerated. Appendix 4 shows that an accelerating area of about 7 sq yd would be needed to produce an ion beam of the necessary 2000 amp and 1500 kw. This result assumes that the accelerating voltage is applied over a distance of only a millimeter. Two fine-mesh wire screens, made with wires of very small diameter, could be placed this far apart and given the requisite potential difference. Thermionic emission from the wires in the outer screen could add electrons to the beam so that the ejected gases and the ship would remain electrically neutral.

It may be remarked that if the accelerating voltage were increased to 100,000 volts, the ion velocities would be about 1000 km per sec, and a space ship could in theory attain a speed of this order after about a hundred years of acceleration. At such a speed about a thousand years would be required to reach the nearest star.

Launching and Use of Ship

The preliminary analysis presented here indicates that there is every reason to believe that an interplanetary space ship could be built with essentially present techniques. Such a ship could not by itself land or take off on any major planet or satellite, although it could readily land on a small asteroid or on one of the tiny moons of Mars, where the small weight of the ship could easily be balanced by the rocket's thrust. However, the ship could proceed from a circular orbit about the Earth to a similar orbit about any other body in the solar system.

How could such a ship be used? Since we are all situated on the Earth's surface, a ship of this type

could be employed only if some means for traveling up to a circular orbit were available. We have already indicated that with the use of chemical rockets this problem is, in principle, solved. One may envisage short-range chemical rockets, with a starting weight of at least several hundred tons, sending a final weight of about a ton up to a circular orbit, where contact would be established with an interplanetary ship.

Probably the most difficult problem of this sort would be the initial launching of the interplanetary ship itself. While a voyage of many hundred million miles in space could readily be achieved by this ship, ascent of the first few hundred miles to a circular orbit would definitely require a booster of some sort. In this case the booster would weigh initially some hundred times as much as the ship, or a few thousand tons. This is definitely a case where the first few hundred miles are the hardest, and in fact the design and construction of such a large "launching rocket" might well be more difficult than the design and construction of the long-range space ship.

Another problem, almost as difficult as that of the take-off, is that of landing. Air resistance seems the obvious way to slow down a ship returning to the Earth's surface. However, a body in a circular orbit has a vast amount of energy, and disposing of all this energy by air resistance without vaporizing the material is no trivial problem. There are a number of ways in which glided flight down from a circular satellite might seem to be feasible. Since this subject is dealt with in another paper in this symposium it will not be discussed here.

One possibility in this connection should be noted, however. Gliders constructed for descending from a satellite will probably require wings and bodies of appreciable weight. It might be cheaper in the long run to fabricate these in space, from the nickel-iron presumably available on asteroids, rather than haul them up to the circular orbit by chemical rockets. An interplanetary space ship could readily land on a small asteroid and might conceivably carry the tools necessary to fabricate out of the iron in the asteroid at least the wings needed; these might then be attached to the rockets shot up from the surface.

Landing on another planet would be more of a problem, since a chemical rocket with large quantities of fuel would presumably be needed for the return trip. Possibly such a rocket could be hauled up from the Earth's surface to the circular orbit without its fuel, and the gaseous fuel collected by long-range spaceships from the atmosphere of various planets—possibly oxygen from the Earth and hydrogen or methane from Titan, the satellite of Saturn. All this material could then be glided down to the surface of Mars. After exploration of the planet, the chemical rocket could be assembled and ascent to a circular orbit made.

Evidently the nuclear-powered electrical rocket could open up many possibilities for interplanetary travel. Only the future can reveal to what extent these possibilities may be realized.

Appendices

1 Power-Weight Ratio

We consider the power consumed in an ideal rocket in which the atoms in the rocket jet have no random motion. Let N atoms per sec, each of mass m , be propelled away from the rocket at a speed v_a . Then the total backward momentum imparted to the atoms per sec will be Nmv_a ; if M is the mass of the ship, and a its acceleration, then

$$Ma = Nmv_a \dots\dots\dots [1]$$

The energy imparted to each atom is $mv_a^2/2$. Hence the power P , in watts, is given by

$$P = \frac{1}{2} Nmv_a^2 \cdot 10^{-7} \dots\dots\dots [2]$$

Combining Equations [1] and [2] we have

$$P = \frac{1}{2} Ma v_a \cdot 10^{-7} \dots\dots\dots [3]$$

Equation [3] gives the power-weight ratio for a given acceleration and atomic velocity v_a . For an acceleration of 0.3 cm/sec^2 , sufficient to attain a velocity of 15 km/sec in two months, and for a value of v_a equal to 100 km/sec , sufficient for a total velocity change of 30 km/sec with a mass ratio of about 1.3, we find

$$\frac{P}{M} = 0.15 \text{ watt/gram} \dots\dots\dots [4]$$

corresponding to a weight-power ratio of 11 lb per hp. Thus, for a ship with a mass of 10 tons, the power capacity must be roughly 2000 hp, or about 1500 kw.

2 Radiation Shielding

We compute the distance at which personnel will be safe from the neutrons produced in a uranium or plutonium pile, generating 1,500,000 watts. With an ideal efficiency of one-half and an actual efficiency of perhaps one-third, the total heat generated will be three times the electrical power produced. Let N_n be the number of neutrons liberated per sec. Since one neutron per fission must go to sustain the reaction,

$$N_n = \frac{3P(y-1)}{10^{-7}E_f} \dots\dots\dots [5]$$

where y is the number of neutrons per fission, and E_f is the energy released per fission, in ergs. While precise values of y are not apparently available, this quantity is somewhat greater than unity. The energy E_f is about $3 \cdot 10^{-4}$. If none of the excess neutrons liberated were absorbed, the flux of neutrons F_n per sq cm at a distance r centimeters would be given by

$$F_n = \frac{N_n}{4\pi r^2} \dots\dots\dots [6]$$

If we combine these equations we find that at a distance of 100 km from a reactor with an electrical power of 1.5×10^6 watts,

$$F_n = 1.3 \cdot 10^2 (y-1) \text{ neutrons/cm}^2 \text{ sec} \dots\dots\dots [7]$$

If $y-1$ is about one, this value of F_n is about the safe

dosage for continuous exposure to fast neutrons.⁴ For comparison, a lethal dose within a short period would be $5 \cdot 10^{11}$ neutrons per sq cm. At the rate obtained from Equation [7] about a hundred years would be required for a dose of this magnitude. It is well known that over a long period of time the human body can stand without any injury many small successive doses of radiation, which would be fatal if received all at once. Consideration of additional neutron absorption within the pile would presumably reduce the neutron flux even further.

The computation of the gamma-ray flux is somewhat more complicated. The lethal dosage of gamma rays corresponds to about the same flux of photons per sq cm as the lethal dose of neutrons, in neutrons per sq cm. Since one would expect the number of gamma rays and neutrons produced in fission to be roughly equal, one may conclude that at 100 km the gamma-ray flux will also be below the safe limit. A more detailed analysis of gamma rays would be necessary for conclusive results, however, especially since the absorption of gamma rays within the pile will probably be less than the absorption of neutrons.

3 Problems of Cooling

The generation of electrical power by means of a heat engine requires that the heat produced at a temperature T_1 be conveyed to a "sink" at a temperature T_2 . The ideal efficiency is $(T_1 - T_2)/T_1$. In interplanetary space the only way heat can be dissipated is by radiation, and the radiating surface A , must be sufficient to radiate at the temperature T_2 the heat flowing to it. If Q represents the heat radiated per sec, measured here in watts,

$$\frac{P}{Q} = z \left(\frac{T_1 - T_2}{T_1} \right) \dots \dots \dots [8]$$

where z is the ratio of the actual efficiency of power generation to the ideal efficiency. The heat radiated per sec from the surface, if we assume this is a perfect absorber, is given by

$$Q = \sigma A_r T_2^4 \dots \dots \dots [9]$$

where

$$\sigma = 5.7 \times 10^{-12} \text{ watts/cm}^2 \text{ deg}^4 \text{ sec} \dots \dots \dots [10]$$

Combining Equations [8] and [9] we obtain for A_r ,

$$A_r = \frac{PT_1}{\sigma z(T_1 - T_2)T_2^4} \dots \dots \dots [11]$$

We shall assume here that T_1 is 900 K (627 C) and T_2 is 450 K (177 C). If we let $z = 2/3$, the over-all efficiency of the heat engine = $1/3$ and, if P is 1,500,000 watts, we find, for the assumed 10-ton ship,

$$A_r = 1.9 \times 10^7 \text{ cm}^2 \dots \dots \dots [12]$$

This would yield a cooling area roughly 30 meters square, radiating on both sides.

⁴ The author is indebted for this figure to L. R. Shepherd, who kindly pointed out an inaccuracy in an earlier version of this paper.

4 Electrical Acceleration of the Propellant

We compute the electrical current required to accelerate a beam of ions for rocket propulsion. The current is related to the power P , in watts, and the potential V in volts by the relationship

$$P = iV \dots \dots \dots [13]$$

The potential V across which the ions are accelerated is determined from the energy equation

$$\frac{eV}{300} = \frac{1}{2} m v_a^2 \dots \dots \dots [14]$$

where e is the charge on each ion, equal to 4.8×10^{-10} electrostatic units if each ion is singly charged, and m is the mass per ion. For nitrogen atoms, m is 2.32×10^{-23} gm, and if we again let v_a equal 10^7 cm/sec, we find

$$V = 730 \text{ volts} \dots \dots \dots [15]$$

Thus, for a power of 1,500,000 watts, we have

$$i = 2000 \text{ am} \dots \dots \dots [16]$$

With so large a current, space-charge limitations must be considered. If the potential drop V occurs across a gap x , the maximum current density, j amp per cm^2 , is given by

$$j = \frac{2^{1/2}}{9\pi x^2 \times 3 \cdot 10^9} \left(\frac{e}{m} \right)^{1/2} \left(\frac{V}{300} \right)^{3/2} \dots \dots \dots [17]$$

The area A_i of the ion source is evidently given by

$$A_i = \frac{i}{j} \dots \dots \dots [18]$$

If we substitute i from Equation [13], for j from Equation [17] and for V from Equation [14], we find

$$A_i = \frac{36\pi x^2 e^2 P 10^7}{m^3 v_a^6} \dots \dots \dots [19]$$

For a power of 1.5×10^6 watts, a velocity v_a of 10^7 cm/sec, an accelerating gap of 0.1 cm, and for nitrogen ions as the propellant, we find

$$A_i = 7.2 \times 10^4 \text{ cm}^2 \dots \dots \dots [20]$$

This area of about 7 square meters required for the ion source is much smaller than the area required for radiative cooling.

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Letters to the Editor*

Boundary Conditions in the Theory of One-Dimensional Flame Propagation¹

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In a discussion of boundary conditions for use in flame speed calculations, Adamson comments on the conditions used by Hirschfelder, and Emmons, and proposes another set himself. A set of differential equations expresses the conditions that have to be met in the interior of a region (of the independent variables), while the boundary conditions define in mathematical form the particular problem whose solution is sought. To state, as Adamson does, that three different boundary conditions are used, is merely to say that three different physical problems are being solved.

Since in each of the three cases the desired information was the flame speed in a given (idealized) combustible mixture, and not the detailed solution of a given flow system, the important question is the relation of the assumed boundary conditions to the resultant flame speed. My work showed that if all possible values of some parameter (say T_0) defining the cold boundary of the combustible stream are chosen, then for the usual ranges of mixture heating value essentially the same flame speed is obtained for essentially all values of the parameter. Thus, in particular, this same plateau value of flame speed is obtained by Hirschfelder's or Adamson's cold boundary condition. In general, a multiply infinite set of other cold boundary conditions will also lead to the same value of the flame speed (same within, e.g., 1 part in 10^6).

From the point of view of the determination of a flame speed for ordinary combustible mixtures then, the cold boundary condition is of little importance.

The cold boundary condition becomes of major importance when a mixture is such that it is barely able to support combustion. The boundary condition is then very important for very rich or very lean mixtures, for very poor fuels, and perhaps for blow-off conditions.

If these questions are being studied, it is clear that a more precise determination of the cold boundary conditions are required. By more precise, I mean a mathe-

matical statement that is related more accurately to the experimental or practical case of interest.

For example, in the solution by Emmons, Harr, and Strong, the premixed combustible was supposed to have flowed together a considerable distance before appreciable temperature changes had occurred. Under these conditions, the heat loss to passage walls can be large compared to the effect of heating from the flame. When such wall-cooling is included, there generally will be a sufficient (but very minute) disturbance of the one-dimensionality of the flow to give the plateau value of flame speed.

When studying the failure of a given mixture to burn or to maintain combustion at a given flame holder, such minute effects may be of critical importance. That considerations such as these are indeed important may be inferred from the fact that combustion test results generally do not scale, i.e., test results obtained on small apparatus cannot be correlated with similar results from large apparatus. This of course means that some of the pertinent variables are unknown and these may well include such things as passage wall properties and ambient conditions as well as the physical and chemical properties of the combustible mixture.

Comments on Adamson's Letter on the Theory of One-Dimensional Flame Propagation

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T. C. Adamson, Jr., has made an interesting analysis² of boundary conditions in the theory of flame propagation on the basis of our earliest publication on this subject. However, since that publication we have done a great deal of work in the development of the theory and have studied the boundary conditions exhaustively. At the present time we are preparing a manuscript on this subject for submission to the *Journal of Chemical Physics* for publication. It is interesting to see that an infinitesimal amount of energy transferred from the flame to the flame holder can serve to stabilize the flame velocity. In this sense the significance of the cold

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¹ Comments on "On the Theory of One-Dimensional Flame Propagation," by T. C. Adamson, Jr., *JOURNAL OF THE AMERICAN ROCKET SOCIETY*, vol. 22, January-February 1952, page 38.

² Gordon McKay Professor of Engineering Science.

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¹ Professor of Physical Chemistry.

² "On the Theory of One-Dimensional Flame Propagation" by T. C. Adamson, Jr., *JOURNAL OF THE AMERICAN ROCKET SOCIETY*, vol. 22, January-February 1952, page 38.

* This section of the *Journal* is open to letters not exceeding 600 words in length (or one and one-half columns) devoted to brief research reports or technical discussions of papers previously published. Such letters are published without editorial review, usually within two months of the date of receipt. The style and manner of submission of letters are the same as for regular contributions. (See inside back cover.)

boundary conditions can be thought of as equivalent to mathematical conditions for stability. In order to give a more detailed comment it would be necessary to present our full analysis, for which the reader is referred to the aforementioned paper.

Resistance to Thermal Shock

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By analysis of the nonsteady thermal stresses developed in a slab of material in unidimensional heat flow, Cheng (1)² obtains as a criterion for the resistance to thermal shock

$$S = \frac{(1 - \nu) \sigma_0}{\alpha E T_0 \sigma_{\max}^*} \dots \dots \dots [1]$$

where failure of the material is predicted for S less than unity. Besides the material properties appearing explicitly in Equation [1] (Poisson's ratio ν , ultimate strength σ_0 , thermal expansion coefficient α , Young's modulus E), the expression for S involves the thermal conductivity k , slab thickness b , and heat transfer coefficient h which enter into σ_{\max}^* , the dimensionless maximum stress in the slab developed under the given conditions of heat transfer, i.e., specified h and source temperature T_0 . For $k/hb > 10$, Cheng finds

$$\frac{1}{\sigma_{\max}^*} \sim \frac{3k}{hb} \dots \dots \dots [2]$$

which, substituted in Equation [1], shows a proportionality between S and k as found in the tests by Lidman and Bobrowsky (2), provided h is assigned the value 50 Btu/ft² hr F or less.

In typical rocket engineering applications, h varies in the range 200 ~ 3000 Btu/ft² hr F, and the parameter k/hb is usually much less than 10, particularly for thick refractories of low thermal conductivity. A detailed study of this regime of k/hb shows in effect a weaker than linear dependence of resistance to thermal shock on the conductivity k , in agreement with data quoted by Norton (3).

It may be noted further that when Equation [1] is applied to a large variety of refractory materials (4) under the conditions $k/hb < 1$, $T_0 > 3000$ F, the above criterion predicts failure which is not in fact observed. This can be readily understood if one considers that the σ_{\max}^* computed on the basis of temperature-independent material properties, heated (or cooled) for a sufficiently long time, may not actually be developed under the particular conditions of the test. For, on the one hand, the elastic behavior postulated in Cheng's derivation is rarely realized at higher temperatures, and some easing of the stresses is bound to occur due to yielding or plastic flow of the material. On the other hand, the times required for the attainment of σ_{\max}^* depend inversely on the heat capacity (per unit volume) of the material, so that, under short-time heat

transfer conditions, the heat capacity also enters as a parameter in the determination of the maximum stresses. These considerations lend emphasis to the author's concluding remarks that the results on thermal shock resistance obtained from tests on special specimens are not sufficient to predict the performance of the material under particular design and heat transfer conditions. Nevertheless, the general correlation of the criterion in Equation [1] with test data strengthens the plausibility of the viewpoint that thermal shock failure is essentially failure under combined thermal and mechanical stresses. From this viewpoint the problem of designing against thermal shock reduces to a problem in stress analysis, in which a straightforward account is taken of the nonsteady thermal stresses associated with nonsteady temperature distributions.

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- 2 "Correlation of Physical Properties of Ceramic Materials with Resistance to Fracture by Thermal Shock," by W. G. Lidman and A. R. Bobrowsky, NACA TN no. 1918, 1949.
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- 4 "Ceramic Materials for Application in the Design of Jet Propelled Devices," by F. P. Saybol, The Ohio State University Research Foundation, Project 252, Report no. 17, Oct. 1946.

Energy from the High Atmosphere

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Several suggestions have been made to the effect that large amounts of energy may be extracted from the higher reaches of the atmosphere. From about 80 km upwards solar energy is stored in photochemical reactions which proceed in the forward direction during sunlight. The reverse reactions which may proceed either during sunlight or darkness return the energy to the atmosphere. These processes include excitation, dissociation, and ionization. In order to determine the possible magnitude of the energy stored in this fashion, several typical examples will be chosen.

The ionized and dissociated properties of the high atmosphere which will be quoted below are those commonly accepted^{2, 3}. In this paper the terminology employed to indicate the higher atmospheric shells and their corresponding approximate altitudes at (45° N or S) are: ionosphere, 80-140 km; mesosphere, 400-1000 km; and exosphere, above 1000 km.

Consider first the energy absorbed per sec in ionizing a typical fraction of air molecules in a unit volume. Assume that 30 ev are absorbed per ionization and that 10³ ion-pairs/sec are produced. On this basis 3 × 10⁴ ev/cm³ sec, or approximately 5 × 10⁻¹⁶ watts/cm³,

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¹ Chief, Ionospheric Laboratory, Geophysics Research Division.

² "The Atmospheres of the Earth and Planets," by G. P. Kuiper, Univ. of Chicago Press, 1949.

³ "The Colloquium on Mesospheric Physics," by N. C. Gerson, Geophys. Res. Paper No. 8, Cambridge Research Center, 1951.

Received Dec. 22, 1951.

¹ Research Engineer, Special Projects Department.

² Numbers in parentheses refer to References at end of letter.

are absorbed. The entrapped solar energy in one cubic kilometer would be about 5 watts, so that to light an ordinary tungsten electric bulb it would be necessary in some fashion to wring out this newly delivered solar energy from 20 cubic kilometers during each second. If this rate of ionization occurred on the average throughout a thickness of one kilometer, the total available power in the sunlit portion of the atmosphere would be about 10^6 kw.

Consider, secondly, the energy available because of the persistent electron density. In the F_2 ionospheric region, the electron density is about $10^6/\text{cm}^3$. With an ionization potential of about 15 ev, the energy available per unit volume if all ions were neutralized in one second is 2.3×10^{-12} watt/ cm^3 , or about 2 kw/ km^3 . If this degree of ionization were found throughout the entire sunlit hemisphere, about 10^9 kw would be potentially available again. (Obviously, once the ionized layers were neutralized, some finite time would be required to restore them to the ionized state.) In this connection a comparison with the solar energy delivered to the top of the atmosphere is desirable. The solar constant is about 2 gram calories/ cm^2 min. If this energy appears on the sunlit portion of the atmosphere at the base of the exosphere, the energy input to the terrestrial atmosphere is about 5×10^{14} kw. However, the exact value of the solar constant is by no means known and may be larger than that given above.

Thirdly, the energy stored in the dissociation of molecular oxygen may be considered. If the maximum number density of atomic oxygen is $10^{12}/\text{cm}^3$, the available energy, considering the dissociation potential of 5 ev, is 8×10^{-7} watts/ cm^3 , assuming complete association in one second. This energy is equivalent to 8×10^8 watts per cubic kilometer, or about 10^{14} kw (in a layer one kilometer in thickness) over the entire sunlit hemisphere.

Finally, consider the energy available to an injected particle because of the high temperature in the ionosphere, about 3900 K at 400 km. The heat transfer to a unit volume of iron originally at a temperature of 300 K would raise the temperature of the steel infinitesimally. If this unit volume of iron passes through 1000 km of air at a number density of $10^{10}/\text{cm}^3$ and at an average temperature of 2500 K, however, the temperature rise in the unit volume of steel would be about 0.01 K (assuming temperature equilibrium between this volume of air and the unit volume of iron). Note that this figure does *not* consider the effect of dynamic heating which under proper circumstances may be considerable.

The examples above err on the conservative side. An ion production of 1000/ cm^3 sec within the F ionic regions may be possible, but this value would be scarcely expected to occur at all zenith angles and throughout an extensive depth in the terrestrial atmosphere. Similarly, the figure of 10^6 ions/ cm^3 for the F_2 layer occurs at noon and decreases to $10^5/\text{cm}^3$ at zenith angles of about 90 deg. Refinements to the conservative estimate chosen would probably not exceed a factor of 10^{-2} . These examples nevertheless indicate

the small amount of energy available from ionization in the high atmosphere.

An example of the utilization of this energy by a rocket may be mentioned. Consider the diameter of a rocket to be about 15 meters, and that in its travel through the ionized region it extracted the ionization energy from all ion pairs encountered. During its traversal of the ionosphere, the rocket would burrow a tunnel of about 10^{12} cm^3/sec at the escape velocity. With a reasonable density of 10^5 ions/ cm^3 in the E layer and an ionization potential of 15 ev, the power available to the rocket from all the ions neutralized is 0.5 watts. In the F layer, the energy would be greater by a factor of 10. At its speed of 11 km/sec, the rocket could traverse a circular orbit at 300 km (the F layer) in about 1 hr, thereby storing in its storage unit 5 watt-hr of energy. (A lead storage battery is usually rated at about 4 kw-hr.) If, however, the rocket traveled at an altitude somewhat above 100 km and completely absorbed the energy of dissociation of molecular oxygen throughout the tunnel it burrowed, the energy available (per sec) would be about 10^3 kw. At this altitude one traversal at a speed of 11 km/sec would thus store 10^3 kw-hr.

The above examples indicate that the stored energy available in the ionosphere in the form of ionized particles is small but that the energy available from dissociation is much larger. In any event, even the energy from the latter source is probably smaller than that which may be obtained from other means.



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Jet Propulsion News

By C. F. WARNER, Purdue University, Associate Editor

with the assistance of W. G. BOHL

Aircraft

THE Glenn L. Martin Company announced that the first of its two experimental XB-51 bombers has been flown to Edwards Air Force Base, Muroc, Calif., to undergo additional flight testing. The three-jet-engine, swept-wing plane has completed its test flight program at Martin Airport, Baltimore.

♦ ♦ ♦

THE Air Force has released a picture (Fig. 1) of the Boeing XB-52 eight-jet-engine heavy bomber, the Stratofortress. The eight Pratt & Whitney J-57 turbojet engines are undergoing engine run-ups at Boeing Field, Seattle, Wash. No further details of the aircraft have been released.

♦ ♦ ♦



FIG. 1 BOEING STRATOFORTRESS SWEEP-WING JET BOMBER

PICTURED in Figs. 2 and 3 is the Republic XF-91 high-altitude interceptor fighter powered by turbojet engine and rocket motors. The XF-91 was developed and built by Republic Aviation Corporation.

♦ ♦ ♦

AN idea of the use of aircraft armament rockets can be obtained from the eighteen 5-in. rockets in place under the wings of the Republic F-84F swept-wing fighter bomber shown in Fig. 4. The F-84F fighter's performance far exceeds that of the Republic's F-84E currently being used by the United Nations in Korea. The F is capable of very high speeds and exceptionally long-range operations. As a ground-support plane, it is designed to carry more armament than the F-84E. The F-84F has a wing span of 34 ft, over-all length of 43 ft, and is 14 ft high at the top of its rudder fin. Its maximum design gross weight at take-off is approximately 25,000 lb.

♦ ♦ ♦

THE first flight of Chance Vought Aircraft's F7U-3 Cutlass, designed to out-fly or out-fight any other carrier-based fighter airplane in the world, has been announced by the Navy. The F7U-3, powered by two

Allison J-35-A-29 turbojet engines, is superior to the F7U-1 in climb, speed, armament, and range. The sweptback-wing tailless fighter has a dual power control system, the first to be installed in any Navy fighter aircraft. Manual controls have been eliminated entirely, and each of the two power systems is completely independent so that neither can fail because of a breakdown in any component of the other system.

♦ ♦ ♦

SUCCESSFUL first flight of the Navy's XA2J-1 North American "Savage" in Los Angeles, January 4, brought to four the number of turboprop-powered aircraft now flying in this country. Two Allison T40 turboprop engines produce more than 11,000 equivalent shaft hp for the XA2J-1. Rated at over 5500 equivalent shaft hp at a weight of 2500 lb, the T40 is the most powerful propeller-type aircraft engine for its size and weight in the world. Designed as a twin-unit engine, the T40 is made up of two identical power sections which are connected by extension drive shafts to a single reduction gear.

♦ ♦ ♦

THE English Gloster Company has developed the world's first twin-jet-engine delta-wing fighter designated the GA5. Powered by two Armstrong Siddeley Sapphire turbojets (7200-lb thrust each), the GA5 is



FIG. 2 REPUBLIC XF-91 ROCKET-ASSISTED JET FIGHTER

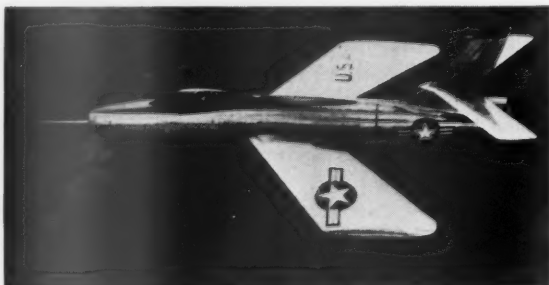


FIG. 3 XF-91 IN FLIGHT



FIG. 4 REPUBLIC F-84F EQUIPPED WITH ROCKETS FOR GROUND SUPPORT

an all-weather day-and-night fighter built to the same specifications as the de Havilland 110.

♦ ♦ ♦

THE Royal Swedish Air Force has accepted deliveries of the Saab-J-29 swept-wing fighter powered by the de Havilland Ghost built under license in Sweden. The top speed of the J-29 is of the order of 650 mph. The Saab works at Linköping has also accepted authorization from the Swedish Air Force to construct a delta wing night fighter, the J-1200, for all-weather operation.

♦ ♦ ♦

A RAMJET-powered research monoplane, the 0.10, developed by Leduc has undergone extensive flight tests in France. Originally air-launched from a Languebec, the plane may soon be launched by a catapult.

♦ ♦ ♦

IT has been reported that Douglas Aircraft Company has completed its supersonic, high-altitude research plane, the X-3, which is powered by a Westinghouse J-40 engine.

♦ ♦ ♦

THE French are experimenting with two new aircraft in the over-600-mph class. One, the SNCASO 6026 Espadon jet fighter, is equipped with a rocket motor installed under the turbojet nozzle. The second plane is the SNCASO M2 Rolls-Royce Derwent-powered fighter.

♦ ♦ ♦

THE Australian Department of Defense is experimenting with both a piloted and a pilotless version of the first all-Australian jet-powered aircraft. The radio-controlled target aircraft and the piloted version have a wing span of under 20 ft, a length of about 22 ft, and are powered by the Armstrong Siddeley Adder 1 turbojet engine. This engine, especially designed for the project, has a ten-stage axial compressor, a two-stage turbine, six combustors, and an OD of 22 inches.

♦ ♦ ♦

OPERATING under a Navy contract, Kaman Aircraft Corporation has installed a 175-hp Boeing XT-50 gas turbine in a Navy K-225 helicopter to conduct a series of flight tests on the new power plant (Fig. 5).

MARCH-APRIL 1952

The turbine is similar to a turboprop installation in an airplane in that the turbine delivers mechanical power to the rotors. Although this turbine has undergone tests in land and water vehicles for some time, this is the first application of the turbine in a helicopter. Kaman engineers predict marked increases in the helicopter's performance as compared to the same machine powered by a conventional piston engine. This increased performance results from the reduction in weight of the power plant. The XT-50 weighs approximately one half as much as the conventional engine and, unlike the piston engine, the turbine requires neither a centrifugal clutch nor a cooling fan.

♦ ♦ ♦

IT is understood that the Air Force is continuing work on its Piasecki XH-16 turbine-powered helicopter. The XH-16 will carry 44 passengers and will eventually be powered by two Allison T-38 turbines.

Turbojet Engines

THE U. S. Navy has selected the Pratt & Whitney Aircraft T34 Turbo Wasp for installation in two Lockheed R70-1's in a program designed to test turboprop-powered transport aircraft. The R70-1 is the Navy version of the new Super Constellation commercial transport. The four T34 engines will replace the four Wright R-3350 compound engines which were originally installed in the R70-1.

An Air Force contract for work on an atomic aircraft engine has been awarded to Pratt & Whitney Aircraft division of United Aircraft Corporation. The U.S. Atomic Energy Commission is co-operating in consideration of the nuclear energy phase of this work. The research department of United Aircraft Corporation participated in the original postwar exploration of the possibilities of utilizing atomic power for aircraft, known as the NEPA project, which was jointly sponsored by the Air Force and the AEC. P & WA will absorb many of the engineers who worked on the NEPA project. Technical supervision of the nuclear engine project will be assigned to Wright A. Parkins, P & WA engineering manager, and Perry W. Pratt will be assistant chief engineer.



FIG. 5 KAMAN HELICOPTER WITH BOEING GAS TURBINE

THE nation's first "jet-propelled laboratory" for high-speed flight testing of turbojet engines of advanced design has been placed in operation by the General Electric Company. The laboratory is a four-jet North American B-45 bomber which is equipped to carry a fifth test engine in a specially designed nacelle under the bomb bay. The bomber was assigned by the U. S. Air Force to the company to flight-test the GE's latest model, the J-73, and the J-47-GE-17 reheat engine. The test engine nacelle is partially retractable into the bomb bay when not in use and will accommodate turbojet engines considerably larger than any yet announced. Many hundred special instruments have been installed to record every phase of the test engine's performance in flight.

* * *

THE French turbojet engine, Atar 101C, now undergoing bench tests at Villaroche, is reported to develop a maximum thrust of 6614 pounds without water injection or reheat.

* * *

AN afterburner is being manufactured by the Solar Aircraft Company for the A. V. Roe Canada "Orenda" jet engine, reported to have a thrust of over 7000 lb.

* * *

A CRACK crew of mechanics at Wright-Patterson Field changed a Lockheed F-80 engine in the record time of 5 min, 50 sec. The feat included dismantling the aircraft, removing the old engine, installing a new engine, installing the aft fuselage, and bringing the new engine up to idling speed.

Test Facilities

TEN new jet-engine test stands, replacing those destroyed in last summer's explosion, will be constructed on a new land site recently acquired by Allison Division, General Motors, near its Plant 5 in Indianapolis, Ind. A 125,000 sq ft building will also be erected for fabrication and assembly of experimental engines.

* * *

USAF will finance part of the cost of modernizing the wind tunnel at Cornell Aeronautical Laboratory so that tests at supersonic speeds may be made.

* * *

SWEDEN, doing extensive research on turbojet engines, has built a unique jet-engine test facility at Trollhatton. A large air chamber has been carved out of rock 280 ft below the level of a river near Trollhatton. During test operations, part of the river is diverted down a tunnel into the chamber, thus compressing the air which is fed through another tunnel into the engine test stands. It is possible to run 40-min engine tests at almost constant pressure with the system. Compressed air supplied by three electrically driven compressors forces the water out of the chamber. Many of Sweden's aircraft manufacturing facilities are underground.

Rocket News

A LONG-RUMORED alliance of Aerojet with two Italian firms for the manufacture of rockets in Italy has been confirmed. Aerojet has combined with FIAT, one of Europe's largest industrial organizations, and Industrie Meridionali Munizioni Affini (IMMA), a supplier of rocket propellants, in establishing an Italian company known as Compagnia Generale Italjet for the purpose of manufacturing and selling rockets to the friendly nations of western Europe. Headquarters of the company are in Rome. Italjet has already received a provisional order for training rockets for the Italian Air Force, and is preparing technical information for a bid on 5-in. rockets.

* * *

THE ONR and OAR are jointly sponsoring an investigation of the potentialities of different acetylenic compounds for use as rocket fuels at New York University's Engineering Research Division.

* * *

IT is understood that the Air Force has been in disagreement with the Army Ordnance Corps in matters concerning the development and purchase of special weapons for combat planes. One of the sore points was the Air Force's negotiation with the American branch of Oerlikon Machine and Tool Works, Buehrle & Co., of Zurich, Switzerland, for the air-to-air rocket manufactured by the Swiss company. The Army has been traditionally opposed to dealing with any independent and privately owned producers of complete weapons. Moreover, the Air Force, under Air Force Regulation 136-1, still has a policy that declares that the service will carry out its development of weapons "within United States industry." Nevertheless, the outstanding performance of certain Oerlikon items has attracted the attention of Air Force technical officers.

Personalities

FRANK L. SNYDER has been named manager of Westinghouse Electric's Gas Turbine Division, succeeding W. B. Anderson who has been appointed assistant to the executive vice-president, defense products, of the Company.

* * *

GARDNER D. CARR has been appointed assistant to the president of Marquardt Aircraft Company. Carr has been director and vice-president of Boeing Airplane Company, vice-president of Glenn L. Martin Company, and executive vice-president of McDonnell Aircraft Company. Carr brings a rich background of production experience to the ramjet power-plant program now under way at Marquardt. Four additional division head assignments have been announced in the Engineering Department. Robert E. Fisher is chief of aerodynamics, Malcolm S. Harned is chief of preliminary design, Robert K. Wead is chief production engineer, and Norman Svendsen is chief of design.

GAYLORD P. HARNWELL, chairman, Department of Physics, and director, Randall Morgan Laboratory of Physics, of the University of Pennsylvania, has been appointed chairman of the Committee on Ordnance of the Research and Development Board, Department of Defense. Dr. Harnwell has been associated with military research and development since 1941.

♦ ♦ ♦

J. B. EDWARDS, project engineer of the DC-6, has left the Douglas Aircraft Corporation to become chief engineer for Hiller Helicopter Company of Palo Alto, Calif.

♦ ♦ ♦

HERBERT A. ELION, formerly project engineer at the M. W. Kellogg Company, has joined the research staff of Paul Rosenberg Associates, consulting physicists, Mount Vernon, N. Y.

Combustion Studies with a Rocket Motor

(Continued from page 85)

sufficient to sustain the wave. This analysis shows that a requirement for high-frequency instabilities, with the second theory, is an injector which will give maximum combustion close to the injector face.

In regard to the question the authors brought out on the complications introduced by the 0.1 lb/sec of nitrogen used to cool the quartz window, we might point out that with a lucite rocket chamber, which was viewed by the same technique at the Lewis laboratory, identical results were obtained. Since this rocket did not have any cool gases flowing past the window, the interpretation was somewhat simplified and, as identical results were obtained, it is very unlikely that the nitrogen produced any serious changes in the combustion process.

Photographic Techniques Applied to Combustion Studies

(Continued from page 91)

the Aerojet organization who helped with this work. Acknowledgments must especially be made to H. L. Coplen and E. R. Petau for the thrust-chamber and injector designs. Operation of the equipment was at various times under the supervision of Test Engineers W. D. Stinnett, S. Herzfeld, and J. C. McConahay. Also, L. Black and W. Koutnik of the Instrumentation Group were very helpful.

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MARCH-APRIL 1952

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American Rocket Society News

By H. K. WILGUS, Associate Editor

Joint ARS-IAS Rocket Propulsion Session

THE papers presented on February 1, 1952, at the technical session on rocket propulsion, cosponsored by the American Rocket Society at the Institute of the Aeronautical Sciences Annual Meeting January 28-February 1, 1952, New York, N. Y., covered a broad range of subjects pertinent to rocket activities. C. W. Chillson presided, following the custom of the IAS of having the president of the collaborating Society act as chairman at such joint sessions. The attendance of over 350 may be considered as indicative of the growing interest in rockets.

The first paper, "Range Formulas for Rocket-Powered Aircraft," by R. W. Allen, described the development of range formulas for two types of rocket paths and compared them with those for an air-breathing engine system following a Breguet path (constant lift-to-drag ratio, constant velocity, and thrust equal to drag). A constant thrust rocket-powered system always gave less range than the air-breathing engine following the Breguet path, while a rocket-powered airplane with a constant thrust phase followed by a glide path was found to have potentially greater ranges than those of the air-breathing engine under certain circumstances. The formulas developed by Mr. Allen and the curves presented are capable of ready extension to cover cases other than those which he considered.

The second paper, "Large Scale Production and Handling of Liquid Hydrogen," by H. L. Coplen, described in considerable detail this interesting project which was undertaken by Aerojet Engineering Corporation with the assistance of H. L. Johnston of the Ohio State University Research Foundation, who acted as consultant. Some 7400 lb of liquid hydrogen was produced with this plant during a period of approximately six months. The schematic flow diagram as well as pictures of the installation and drawings of critical elements were included, together with descriptions of some of the special requirements and particular precautions necessary to produce this liquid, boiling at only 34 F above absolute zero. Based on the experience gained with this unit, a larger unit was designed with a capacity of approximately 3000 lb per day. While the first unit employed a high pressure system (approximately 2000 psig) and obtained cooling by the Joule-Thomson effect as the hydrogen was expanded through throttling valves, it was found desirable in the larger unit to specify a low-pressure (approximately 220 psig) closed-cycle

helium refrigeration system employing three turbo expanders to extract work from the refrigerant at steadily decreasing temperature levels. This refrigeration system would be used to condense gaseous hydrogen at nearly atmospheric pressure. In addition to discussing the production phase, Mr. Coplen also covered requirements for storage and handling. He discussed also the materials of construction, the problems of insulation, and presented an analysis of storage losses including those associated with the exothermic ortho-para conversion of hydrogen. He also commented on safety precautions which he indicated on the whole to be less rigorous for liquid hydrogen than for high-pressure gaseous hydrogen. In general, this paper gave a very complete picture and was refreshingly free from security deletions.

The final paper by Paul F. Winternitz on "The Role of Research in Rocket Development" traced the history of research contributions to the rocket field and took a long-range look at the problems of the future and the fields of research potentially involved. Dr. Winternitz postulated the ultimate problem as being one of the expansion of inhabitable space, with the rocket being the only power plant capable of transporting man and his effects to other planets or to man-made

satellites. He emphasized the delays in the transmission of fundamental knowledge from the research worker to the engineer, stating that "this is...for progress, the rate-determining step," and pointed out that "in fact, quite efficient rockets might have been built with the knowledge available fifty years ago."

Preprints of the above papers are available from the Secretary of the Society at \$0.50 per copy for nonmembers; \$0.25 per copy for members.

B.I.S. Journal Available to ARS Members

THE American Rocket Society currently receives from the British Interplanetary Society, on an exchange basis, a limited number of copies of its *Journal*. This publication specializes in articles and discussions on a professional plane covering many aspects of the speculative field of space travel. Since insufficient copies are received to permit distribution to all ARS members, and since some members are more interested than others in this phase of rocket technology, the Board of Directors has decided to distribute these *Journals* upon request, within the limit of copies available. A handling fee of \$1.00 is charged for a subscription to all six 1952 issues. Address your request, together with check or money order, to: Secretary, American Rocket Society, 29 West 39th Street, New York 18, N. Y.

Netherlands Society for Space Travel Founded

ON December 21, 1951, the Nederlandse Vereniging voor Ruimtevaart (Netherlands Society for Space Travel) was founded at Dordrecht, Holland.

This Society is devoted to the promotion of interest in and the study of space travel and rocket engineering in all their aspects. For that purpose the Society will publish papers, organize meetings, exhibitions, and film presentations.

Foreign members will also be accepted. As a tentative arrangement, the Society will be an affiliate of the Royal Netherlands Aero Club with the exceptional regulation that members of the Netherlands Society for Meteorology and Astronomy (Nederlandse Vereniging voor Weer-en Sterrenkunde) may also become members of the Netherlands Society for Space Travel.

The provisional Council is composed of the following: *Chairman*, J. M. J. Kooy;



PAUL F. WINTERNITZ
Director of research, Reaction Motors, Inc., whose paper was presented at the IAS Annual Meeting, Feb. 1, 1952.

Vice-Chairman, J. J. Raimond, Jr.; Secretary, J. de Groot; Council Members, J. Geertsma, J. H. Houtman, G. de Koningh, and J. v. d. Vliet.

Those who are interested are invited to write for further information to the Secretary, whose address is Anna Paulownaplein 3, The Hague, Holland.

The First International Astronautical Congress (1950)¹

By Frederick I. Ordway

THE First International Astronautical Congress took place in Paris from September 30 to October 2, 1950, in the Grand Amphitheater of the Sorbonne. It was attended by a large audience, which included the reviewer (in a very unofficial manner—he was a student at the Sorbonne at the time). The most important session, open to an enthusiastic public, occurred on the 30th of September, while the remaining two days were reserved for closed committee gatherings.

Alexandre Ananoff, director of the Groupement Astronautique Français, organized, almost single-handed, this historic Premier Congrès International d'Astronautique. Representatives from France, England, Germany, Austria, Denmark, Spain, and Argentina attended and gave addresses. The over-all character of the meeting was general. The speeches concerned mainly the Congress itself and the fact that astronautics experts and delegates from many nations had gathered together. More technical sessions would be reserved for future international reunions, the importance of which was agreed by all to be great. It was regretted that the United States was not officially represented and the opinion was that since the U. S. does most of the practical work on rockets, it had little to gain from such a meeting. However, the American Rocket Society has since joined the International Astronautical Federation, and was represented at the Second International Congress in London, so our position has been clarified.

Following the discussions, four movies were shown, including "The Girl in the Moon" (made in 1928 under Hermann Oberth's scientific direction), "Rocket Experiences at Raketenflugplatz" (produced by Rudolf Nebel), "V-2 Firings at White Sands," and "Destination Moon."

The honorary president of the Congress was Henri Mineur, director of the Institut d'Astrophysique; the president was Alexandre Ananoff, director of the Groupement Astronautique Français. There was also a distinguished committee of

honor of French scientists. The official representation was as follows: *France*: H. Mineur, Mme. de Vendevure, A. Ananoff; *England*: A. V. Cleaver, president of the British Interplanetary Society; *Germany*: Mr. Gartmann, scientific director of the Gesellschaft für Weltraumforschung; *Argentina*: T. Tabanera, president of the Sociedad Argentina Inter-Planetaria; *Austria*: F. Cap, president of the Oesterreichische Gesellschaft für Weltraumforschung; *Denmark*: Mr. Hansen, vice-president of the Dansk Selskab for Rumfarts-Forskning; *Spain*: T. Múr, president of the Asociacion Espanola de Astronautica.

A special astronautical exhibition was held concurrently at the Palais de la Decouverte in honor of the Congress.

Southern California Section Meetings

THE Board of Directors of the ARS Southern California Section met on January 4, 1952, at the Pasadena Athletic Club, Pasadena, Calif., to review the four successful meetings held in 1951, and to initiate plans for activities in 1952.

Following the reports of R. J. Lodge, secretary-treasurer for 1951, and B. L. Dorman, program chairman and 1952 president, G. L. Sutton, winner of the G. Edward Pendray Award for his book, "Rocket Propulsion Elements," presented some highlights of the ARS National Convention.

Plans for 1952 meetings include sessions on solid propellants, ramjets, and pulse-jets, with emphasis on the continuance of a high technical level of membership.

ARS Convention Papers Read

On January 24, 1952, at a dinner meeting held at the Pasadena Athletic Club, Pasadena, Calif., 200 Southern California Section members and guests were given an opportunity to hear two of the papers that

had been presented at the ARS National Convention.

The multimillion-dollar rocket test facilities now under construction at Edwards Air Force Base, Calif., was the subject of the paper by R. A. Schmidt and D. L. Dynes. Capt. Schmidt described the station as one designed to offer great versatility in propellant utilization, thrust ratings, engine mountings, and instrumentation. The engines to be tested and developed there, he said, will include rocket engines for missiles, aircraft, and assisted take-off units.

As an added attraction in connection with the second paper, "Photographic Techniques Applied to Combustion Studies in a Two-Dimensional Transparent Thrust Chamber," by J. H. Altseimer, of Aerojet Engineering Corporation, high-speed films in black-and-white and in color were shown of a transparent rocket motor in action. Mr. Altseimer described the facilities and techniques employed and pointed out what was perhaps the most interesting visual phenomenon—the appearance of cylindrical flame fronts parallel to the axis of the combustion chamber. (See pages 86-91 in this issue for the paper printed in full.)

1951 Student Award Presented

The student paper adjudged best by the ARS Awards Committee for 1951 was "Design and Construction of a Hydrogen Peroxide Solid Catalyst Rocket," by David G. Elliott and Lee Rosenthal, both at the California Institute of Technology. Because the recipients had been unable to be present at the ARS National Convention to accept the Award, a special ceremony for the presentation was a feature of the Dinner Meeting.

Mr. Elliott accepted the Award from C. C. Ross, member of the ARS National Board of Directors, who made the presentation on behalf of past-president H. R. J. Groesch.



MR. W. J. CECKA, JR., VICE-PRESIDENT, SOUTHERN CALIFORNIA SECTION, ADDRESSING THE SECTION'S DINNER MEETING HELD AT THE PASADENA ATHLETIC CLUB ON JANUARY 24.

¹ EDITOR'S NOTE: This event has not been reported previously in the JOURNAL. In view of the recent affiliation of the ARS with the International Astronautical Federation, it seems appropriate to complete the record with this historical account of the origin of the international idea.



1951 STUDENT AWARD PRESENTED

Mr. Chandler C. Ross presenting the 1951 Student Award to David G. Elliott (left) at the Southern California Section Dinner Meeting, January 24.

The paper describes how the students designed, constructed, and static-tested a liquid-propellant rocket-powered test vehicle, finally firing it successfully in an area in a California desert.

COPIES OF PAPER NOW AVAILABLE

Copies of this paper with illustrations may be purchased from the Secretary, American Rocket Society, 29 West 39th Street, New York 18, N. Y. Order Reprint No. 54-52; prices: \$0.25 to members, \$0.50 to nonmembers.

New York Section Hears Golay on Space Travel

THE feature of the monthly meeting of the New York Section held on Friday evening, January 18, 1952, at the Engineering Societies Building, 29 West 39th St., New York, N. Y., was the address by Marcel J. E. Golay on "Radio Ranging in Outer Space."

W. Mount first opened the meeting by introducing the Section's new president, Lee J. Bregenzner, of the Port of New York Authority. Mr. Bregenzner then presented the speaker of the evening, Dr. Golay, chief scientist, Squier Signal Corps Laboratory, Fort Monmouth, N. J. Dr. Golay, a member of the Signal Corps staff for twenty years, is credited with several inventions used during World War II in sound-ranging sets against heavy military artillery and in the hydracoustic defenses of Puget Sound, Wash.

Dr. Golay described some systems of measurement into outer space and the establishment of a communications system to guide space ships of the future. The task of determining the position and velocity of a manned rocket within our solar system, he said, may prove to be one of the easiest of the major tasks imposed by interplanetary travel.

Present knowledge of radio propagation and electronic techniques, he pointed out, permits forecasting of the lines along which interplanetary instruments should be developed, and of the accuracy to be expected from them.

At the Section's second monthly meeting held on February 15, at the Engineering Societies Building, members heard T. F. Reinhardt of the U. S. Naval Rocket Test Station, Lake Denmark, N. J., speak on "Unusual Applications of the Momentum Principle," the subject of his paper which was presented at the ARS National Convention in Atlantic City, N. J.

Mr. Reinhardt discussed the principles governing schemes of propulsion by extremely high energy particles, and the engineering problems that are encountered in their application. He pointed out the practical possibility of using a working fluid which receives its energy from another power source such as a nuclear reactor, and compared various working fluids.

New ARS Groups Under Way

Northeastern New York Section. In Schenectady, N. Y., a group has been working on the establishment of the Northeastern New York Section and has already prepared the first draft of its By-Laws. One of the initial activities was the formation of biweekly classes in rocket techniques. These classes, under the instruction of General Electric Company engineers, are expected to attract many new members and to contribute toward making the Section a well-established and active one.

At a meeting on February 12, 1952, Kurt Berman and Stanley E. Logan, General Electric Company, presented their paper, "Combustion Studies with a Rocket Motor Having a Full-Length Observation Window," accompanied by a moving picture showing the development of the V-2 rocket. (See pages 78-85 in this issue for the paper printed in full.)

Washington-Baltimore Section. The first membership meeting of this newly formed Section will take place on March 21, 1952, at the U. S. Dept. of Commerce Auditorium, Washington, D. C. Wernher von Braun, technical director, Guided Missile Development Group, Redstone Arsenal, Ala., will address the members, and a panel of experts will be led by Milton Rosen of the Naval Research Laboratory. A full report of this meeting is to be given in the May-June issue of the JOURNAL.

Plans to develop this Section into a well-organized technical and scientific group are well under way. Technical sessions and general meetings have already been scheduled to include such speakers as Fritz Zwicky, Theodore von Kármán, and Secretary of the Navy, Dan A. Kimball.



ARS HONORARY CHARTER MEMBERSHIP

(Left to right): Dr. Wernher von Braun, Dr. R. W. Porter, and Col. C. C. Hudson, commanding officer at Redstone Arsenal, who is receiving an honorary charter membership in the ARS at the organizational meeting of the Redstone Section on January 24.

Redstone Section. The beginning of a membership drive to organize the Redstone Section opened auspiciously on January 24, 1952, at the Russel Erskine Hotel, Huntsville, Ala. Over 200 Redstone Arsenal scientists and technicians attended the Dinner Meeting with the purpose of making the new Section a pilot chapter for other ARS organizations in that part of the country. The chairman for the evening was H. L. Thackwell, Jr., president of the Section.

After the dinner an overflow of interested persons resulted in an attendance of over 400 at the organizational meeting which followed. Aiming at a goal of 500 membership, the acting officers had arranged a special program for the evening.

Colonel Carroll C. Hudson, commanding officer at Redstone Arsenal, was presented with an honorary charter membership in the American Rocket Society. This membership was given in recognition of his interest in the organization and for his contributions to the missile field. R. W. Porter, member of the ARS National Board of Directors, made the presentation.

The two main speakers for the evening were R. W. Porter and Wernher von Braun, technical director, Guided Missile Development Group at Redstone Arsenal.

Dr. Porter, who is in charge of the Army Guided Missile Project at the General Electric Company, gave an over-all picture in his talk, "American Experience with the V-2," of the difficulties encountered in the experiments with the rocket. Dr. von Braun then followed with an interesting and detailed comparison of Dr. Porter's experiences with his own in Germany with the V-2 rocket.

Col. Hudson concluded the meeting with a question-and-answer period setting forth the aims of the American Rocket Society's new Section and what it can accomplish for technical personnel at the Redstone Arsenal and throughout the Southeast territory.

Technical Literature Digest

By H. S. SEIFERT, California Institute of Technology, Associate Editor

CONTRIBUTORS: D. Altman
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H. S. Seifert
F. H. Wright

Book Reviews

THE AURORAE, by L. Harang, John Wiley & Sons, New York, 1951, x + 166 pp. \$4.50.

Reviewed by W. H. PICKERING
California Institute of Technology

This book, the first of a series to be published by John Wiley & Sons as the "International Astrophysics Series," is an interesting collection of experimental facts concerning the aurora and experimental methods for the study of aurorae. The author, Dr. Harang, is chief scientist at the Norwegian Defense Research Establishment. He has been active in auroral investigations for many years, and, as one of the group of Norwegian scientists from whom most of our experimental data on the aurorae are derived, he is well fitted to write on this subject.

The book contains an excellent description of the photographic and other techniques used for studying the appearance, spatial location, and spectra of the aurora. The results of numerous investigations are referred to and briefly summarized.

The relations between the aurora, magnetic storms, and ionospheric storms are presented briefly from the experimental point of view. There is little discussion of the solar origins of these phenomena. A chapter is devoted to the corpuscular theory of the aurora. Although some difficulties with this theory are noted, there is no mention of the ultraviolet light theory of Hulbert and Maris.

A minor comment on the translation is that the term "energy level" of an atom or molecule is occasionally rendered "energy niveau."

The references appear to be generally adequate, although it is surprising not to find Mitra's recent book, "The Upper Atmosphere," among them.

Students of the upper atmosphere and of terrestrial electric and magnetic phenomena will find this volume a useful addition to their bookshelves. Additional volumes of this series listed as "in preparation" are entitled "Comets," "Interstellar Matter,"

"Astronomical Photometry," and "The Earth and the Planets."

L'ASTRONAUTIQUE, by Alexandre Ananoff, Librairie Arthème Fayard, 18-20 Rue du Saint-Gothard, Paris XIV, 1950, 498 pp.

Reviewed by FREDERICK I. ORDWAY
Reaction Motors, Inc.

Mr. Ananoff, a hard and enthusiastic worker in the field of astronautics, has produced an excellent book. It is up to date, well organized, and contains more than 150 valuable tables, graphs, schematics, and illustrations.

The book logically begins with the distant origins of astronautics. Everything from "badly defined mysterious substances" and light pressure to antigravitational material and volcanos projected the early space ship. After a quick scan of these devices one is led immediately into the substance of the book. After a survey of such topics as the bases of astronautics, the role of the planets' atmospheres, laws of gravitation, and escape velocities, the reader is introduced to the rocket itself.

The chapters on the rocket motor, while wide in coverage, are weak in detail. All types of systems are considered, from powder rockets to atomic propulsion, but not enough attention is given to the actual working of the motors.

Ananoff carefully takes up the German wartime and American postwar rocket projects. The V-2 is discussed in detail, along with the never-completed A-10 intercontinental rocket, which was supposed to travel over 5000 km per hr. In this connection Sanger's and Bredt's short-wing antipodal bomber was introduced. This rocket was supposed to "bounce off the dense layers of the atmosphere," similar to a stone skipping across a lake. It would be catapulted into initial flight, and would fly at some 7000 km per hr.

Of great historical interest is the presentation of the schemes for satellite vehicles or stations by such men as Ziolkowsky (the cosmic city), Noordung (a 3-component affair including an inhabited wheel, an

observatory, and machine house), and Oberth. More recent satellite vehicle plans are also presented.

The step rockets provide interesting material for further chapters. Enormous units, having up to five stages and final velocities of 8950 meters per sec, are proposed. Ananoff's conception of the confined life aboard the final stage of such a rocket makes interesting reading. How would the air be replenished? What about the CO₂? Water vapor? Heat control? Light and food? A section on *la cuisine cosmique* answers this last question in a typical French manner.

Factors concerning piloting and control of the space ship, trajectories, free fall, routes to the interior and exterior planets, plotting position in space, relativity and astronautics, effect of acceleration on human resistance, space suits, and landing on the planets are discussed in their turn. The author, indicating his anxiety over a return to the earth, introduces several methods of braking, including the parachute, braking by atmosphere resistance, tangential approaches, and concentric ellipses.

"L'Astronautique" is one of the few good, most nearly complete books in any language dealing with rockets and astronautics, and can be profitably read by the technical and nontechnical man alike. It is unfortunate there is no English translation at the present time.

L'ASTRONAUTIQUE, by Lionel Laming, Presses Universitaires de France, 108, Boulevard Saint-Germain, Paris, 1950, 110 pp. with glossary and bibliography.

Reviewed by FREDERICK I. ORDWAY
Reaction Motors, Inc.

This short, nontechnical book introduces the reader to astronautics by giving a concise, easily understood discussion on rockets and propellants, as well as information on the more important missiles developed both here and abroad. About two-thirds of the book is concerned with such topics as atomic energy as applied to rockets, interplanetary navigation, and the physiological problems of man under conditions of both high acceleration and weightlessness. An interesting table on the animal experiments of Rynin is included. Mice, rats, rabbits, and other animals were subjected to speeds of rotation for certain lengths of time. Some of the creatures perished while others suffered varying

EDITOR'S NOTE: This collection of references is not intended to be comprehensive, but is rather a selection of the most significant and stimulating papers which have come to the attention of the contributors. The readers will understand that a considerable body of literature is unavailable because of security restrictions. We invite contributions to this department of references which have not come to our attention, as well as comment on how the department may better serve its function of providing leads to the jet propulsion applications of many diverse fields of knowledge.

losses of co-ordination. M. Laming asserts that some monkeys had undergone a *centrifugation experimentale* of 27 g without perishing, and adds that "littleness favors resistance, since the supporting surface diminishes less quickly than the weight."

A chapter on planetary exploration and applications of astronautics completes this well-written, handy little book, well designed to acquaint the reader with the approaches of the rising science of astronautics.

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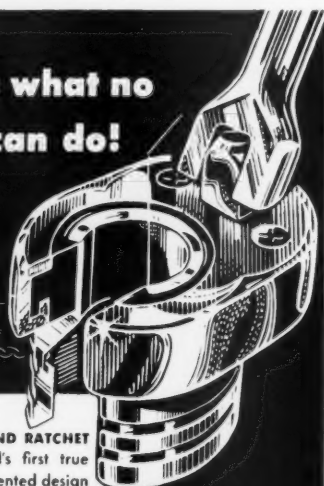
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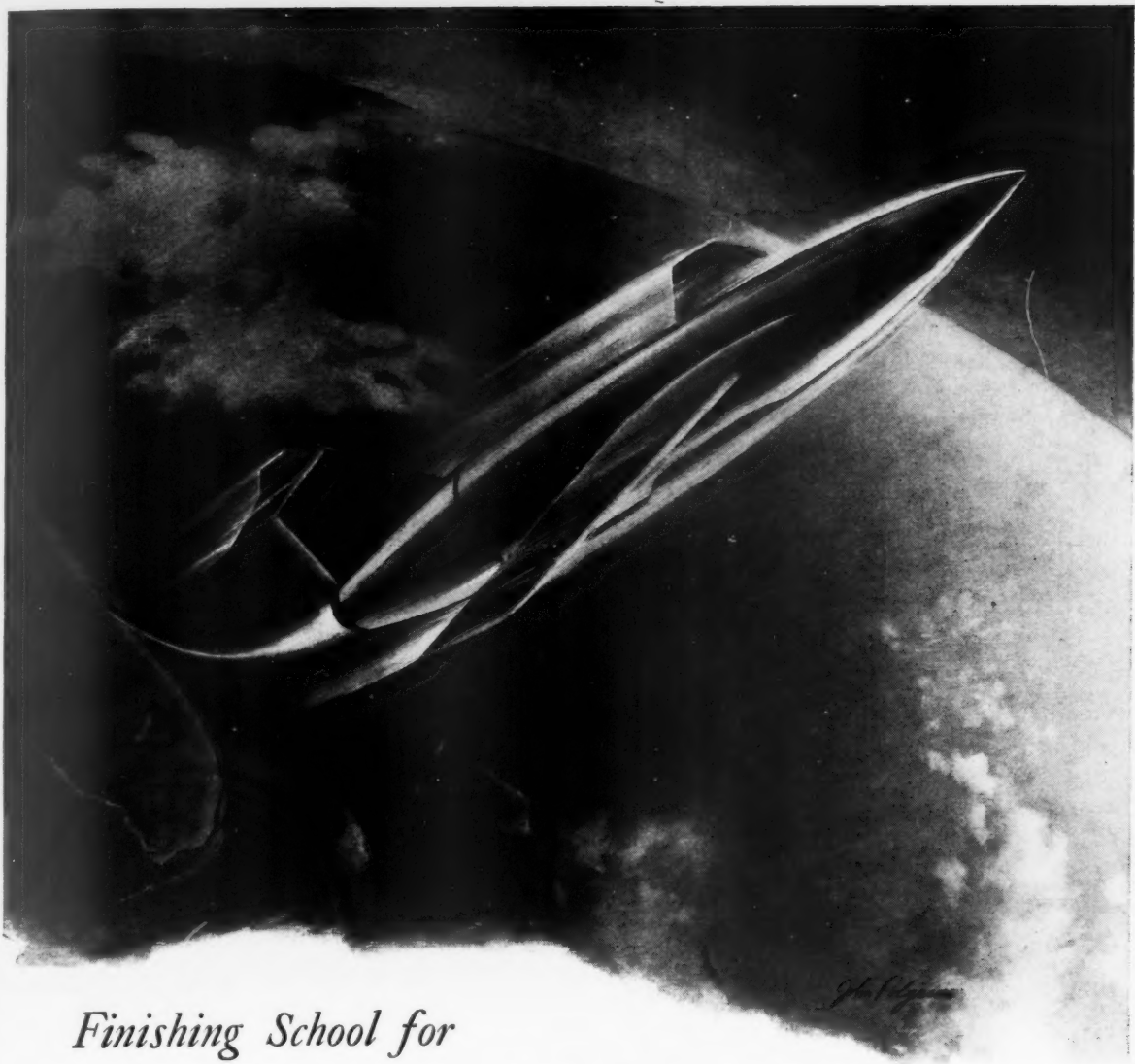
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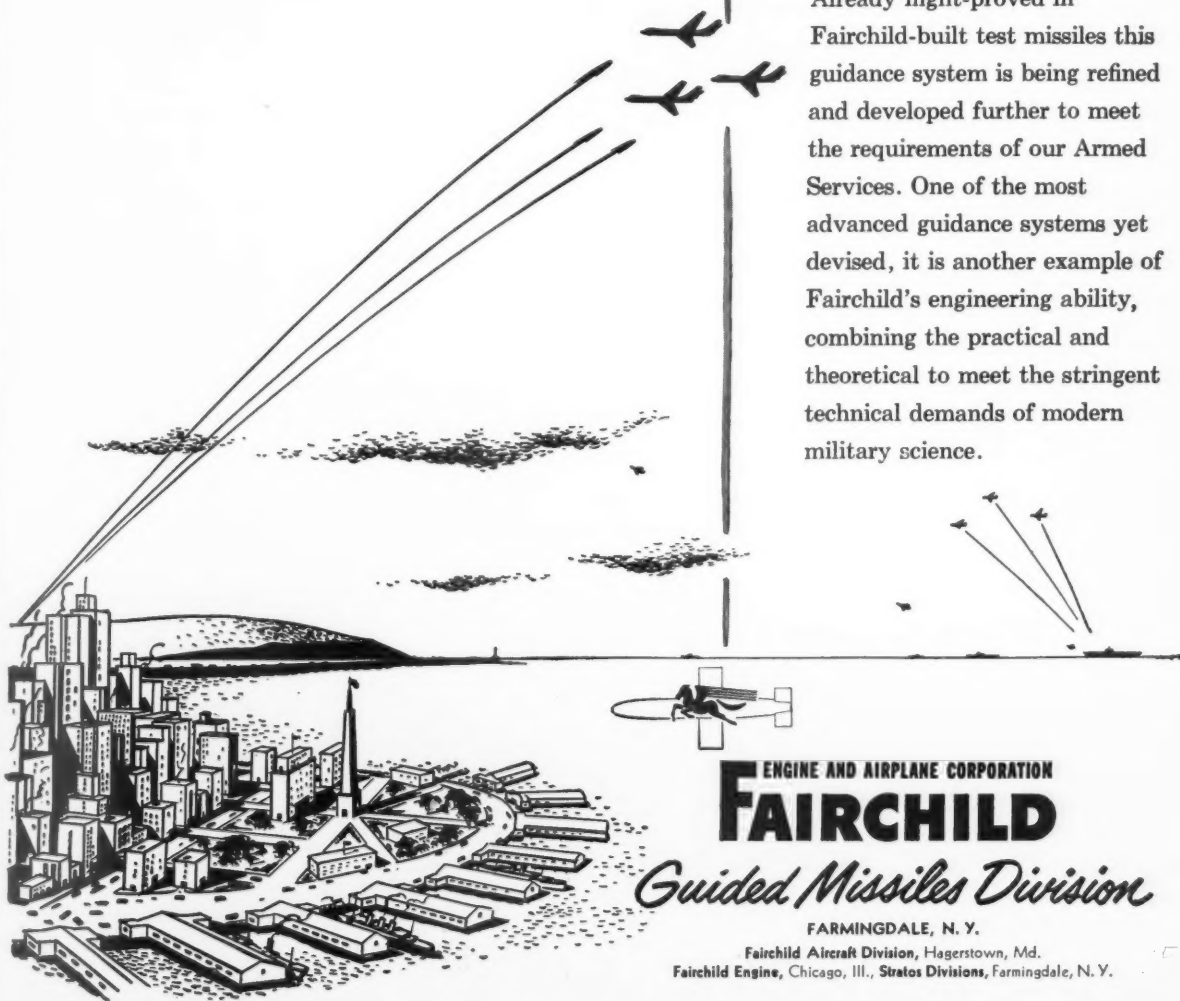
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Ferrous Fluoborate
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Lead Fluoborate
Nickel Fluoborate
Silver Fluoborate
Stannous (Tin) Fluoborate

NON-METAL FLUORIDES

Boron Fluoride Gas
Boron Fluoride-Diethyl Ether
Complex
Boron Fluoride-Phenol Complex
Boron Fluoride-Other Complexes
Sulfur Hexafluoride

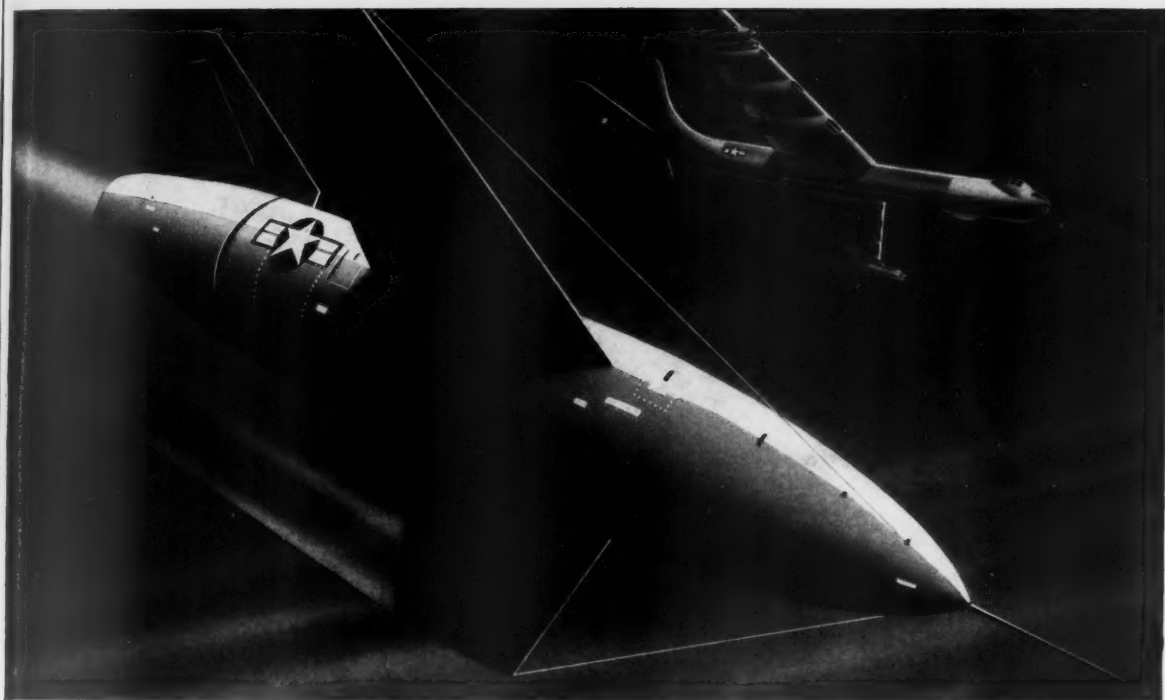
HALOGEN FLUORIDES

Bromine Trifluoride
Chlorine Trifluoride
Iodine Pentafluoride

ORGANIC FLUORINE COMPOUNDS

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Genetron 160—CHCl=CF₂
Genetron 170—CCl₂=CF₂
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*The products listed include those which are commercially available as well as a few presently produced only in experimental quantities. For further information on any of these, or on other fluorine compounds you may require, consult the General Chemical Product Development Department.



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